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ADVANCED PROP-FAN ENGINE TECHNOLOGY (APET) SINGLE- AND COUNTER-ROTATION GEARBOX/PITCH CHANGE MECHANISM

FINAL REPORT
by

C. N. Reynolds

COMMERCIAL ENGINEERING
PRATT & WHITNEY
UNITED TECHNOLOGIES CORPORATION

PREPARED FOR:

NASA
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Space Administration

Lewis Research Center
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44135

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16. ABSTRACT

Volume I reports the preliminary design of advanced technology (1992) turboprop engines for single-rotation Prop-Fans, the conceptual design of the entire propulsion system, and an aircraft evaluation of the resultant designs.

Four engine configurations were examined. A two-spool engine with all axial compressors and a three-spool engine with axial/centrifugal compressors were selected. Integrated propulsion systems were designed in conjuntion with airframe manufacturers. The design efforts resulted in 12,000 shaft horsepower engines installed in over the wing installations with in-line and offset gearboxes.

The Prop-Fan powered aircraft used 21 percent less fuel and cost 10 percent less to operate than a similar aircraft powered by turbofan engines with comparable technology. All emission and acoustic regulations projected for the early 1990's were met. A comprehensive technology plan will address key engine and propulsion system technologies

Volume II reports the preliminary design of advanced technology (1992) gearboxes and pitch change mechanisms for single- and counter-rotation Prop-Fan applications.

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FOREWORD

This report presents the results of a definition study conducted to: (1) identify promising propulsion systems for a Prop-Fan powered aircraft, (2) evaluate the propulsion systems in a short-range airplane, and (3) prepare a comprehensive program for verifying key engine technology components by 1988 in order to ensure certification of a Prop-Fan powered aircraft by 1992. This study was conducted as part of the Advanced Prop-Fan Engine Technology Program under Contract NAS3-23045.

The NASA Program Manager for this contract was Mr. G. A. Kraft of the Propulsion Systems Division, Lewis Research Center, Cleveland, Ohio. The Pratt and Whitney Program Manager was Mr. C. N. Reynolds. The principal technical contributors were Messrs. Joel Godston, Wade Ferguson, Robert Owens, Dudo Tari, and John Kiraly.

Pratt & Whitney would also like to acknowledge the technical contributions of the four airframe manufacturers participating in the APET Definition Study: Boeing, McDonnell Douglas, Lockheed-Georgia, and Lockheed-California.

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SUMMARY

SECTION 1.0
SUMMARY

The Advanced Prop-Fan Engine Technology (APET) Definition Study results show that a Prop-Fan powered aircraft can provide a 21% improvement in fuel burn and a 10% advantage in direct operating costs relative to a turbofan powered aircraft with comparable technology. While the Prop-Fan powered aircraft demonstrated significant advantages over a comparable turbofan, several key technologies must be verified before industry will commit to full-scale development leading to certification. Key engine-related technologies include the large horsepower size reduction gearbox, Prop-Fan/nacelle/inlet/compressor interactions, and the small size high-pressure compressor. Additional studies must also be conducted with major airframe manufacturers to address key issues related to engine/aircraft integration.

To initiate the APET program, Pratt & Whitney prepared a Study Procedures and Assumptions document to define the reference aircraft, aircraft mission, reference turbofan engine, fuel burn and direct operating cost trade factors, and other key ground rules for the study. The document was reviewed by the four major airframe manufacturers participating in the APET program; Boeing, McDonnell-Douglas, Lockheed-California and Lockheed-Georgia. The final document reflects their comments and suggestions.

To examine a variety of approaches to turboprop propulsion, four different engine configurations were evaluated; (1) a two-spool engine with axial compression, (2) a three-spool engine with axial/centrifugal compression, (3) a "reversed" three-spool axial/centrifugal engine, with the inlet at the rear and the turbine in front, and (4) a three-spool engine with axial compression. An optimum cycle for advanced turboprop engines was defined using trade factors developed in Task I. Using this optimum cycle, the engines were evaluated on the basis of mechanical design, performance, design assurance issues, and environmental considerations. The two-spool engine with axial compression and the three-spool engine with axial/centrifugal compression demonstrated the best combination of fuel burned and direct operating costs and were selected for further evaluation.

Conceptual designs of integrated propulsion systems for the two engine configurations, including Prop-Fan, reduction gear, and nacelle, were submitted to the airframe manufacturers for critique and comment. Using their input, Pratt & Whitney, with the concurrence of the NASA Program Manager, selected two final propulsion system configurations for further evaluation. Key features of these systems include: (1) selection of a 12,000 shaft horsepower base engine size for the two-spool and three-spool configurations; (2) the option of using an in-line or offset reduction gear; (3) use of an over-the-wing installation to minimize landing gear length.

The integrated turboprop propulsion systems were then evaluated in a reference 120-passenger aircraft over a typical mission. The Prop-Fan powered aircraft demonstrated a 21% fuel burned improvement and a 10% advantage in direct operating cost over a turbofan powered aircraft with comparable technology. The Prop-Fan powered aircraft is also expected to meet all emissions and flyover acoustic regulations projected for the early 1990's.

In order to achieve the goal set for the APET program - certification of a Prop-Fan powered aircraft by 1992 - a comprehensive engine technology program has been developed. This program provides detailed verification plans for key engine-related technologies and identifies major technical considerations which should be addressed in engine/aircraft integration studies. It is recommended that full support be given to this program to ensure verification of critical technologies by 1988 and certification of a Prop-Fan powered aircraft by 1992.

SECTION 2.0
INTRODUCTION

SECTION 2.0
INTRODUCTION

Previous studies conducted by NASA and Pratt & Whitney indicated that a new high-speed propeller, the Prop-Fan, coupled with an advanced turboprop engine, could play a significant role in reducing the fuel consumption and operating costs of aircraft scheduled for service in the 1990's and beyond. The most promising first application for Prop-Fan propulsion is in short/medium range 100-120 passenger aircraft; studies indicate that more than half of the existing fleet must be replaced with more fuel efficient airplanes in the early 1990's. Introduction of a viable Prop-Fan propulsion system for these aircraft could save billions of gallons of fuel over the life of these aircraft.

The objectives of the Advanced Prop-Fan Technology (APET) Definition Study were to: (1) identify promising propulsion systems for a Prop-Fan powered aircraft, (2) evaluate the propulsion systems in a short-range aircraft, and (3) prepare a comprehensive program for verifying key engine technology components by 1988 in order to ensure certification of a Prop-Fan powered aircraft by 1992. The APET study program consisted of five technical tasks.

Task I - Study Procedures and Assumptions

Pratt & Whitney prepared a Study Procedures and Assumptions document to define the ground rules for the APET program. Aircraft related issues were reviewed by four major airframe manufacturers (Boeing, McDonnell Douglas, Lockheed-Georgia, and Lockheed-California). Hamilton Standard provided information on the Prop-Fan.

Task II - Engine Configuration and Cycle Evaluation

Four different turboprop engine configurations were evaluated. Trade factors from Task I were used to define the optimum engine cycle and to select the most promising candidates for further study.

Task III - Propulsion System Integration Studies

The most promising engine configurations were assessed in integrated propulsion systems which included the Prop-Fan, reduction gear, and nacelle. Propulsion system designs were reviewed by the airframe manufacturers. Data packages and computer decks were prepared for each integrated propulsion system.

Task IV - Engine/Aircraft Evaluation

A mission simulation study was conducted to assess the merits of the turboprop propulsion systems relative to a comparable turbofan propulsion system.

Task V - Advanced Prop-Fan Engine Technology Plan

Pratt & Whitney identified the key engine technologies required for an advanced Prop-Fan propulsion system and prepared a comprehensive program for technology verification.

Section 3.0 of this report summarizes the key results of the APET Definition Study. Section 4.0 contains a detailed discussion of the results. Conclusions and recommendations are presented in Section 5.0. The Propulsion System Integration Package, which will facilitate future evaluations of the Prop-Fan propulsion system, has been supplied to NASA and the airframe manufacturers.

Proprietary information, including engine technology verification program costs and engine and gearbox acquisition and maintenance cost data, is presented in a separate volume.

SECTION 3.0
SUMMARY OF RESULTS

SECTION 3.0
SUMMARY OF RESULTS

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SECTION 3.0
SUMMARY OF RESULTS

3.1 INTRODUCTION

The Advanced Prop-Fan Engine Technology (APET) study has identified candidate Prop-Fan systems, evaluated them in a short-range aircraft, and prepared a key engine technology verification plan for the APET program.

The selected Prop-Fan propulsion system was compared with a similar technology turbofan propulsion system on the basis of fuel burned and direct operating cost; both systems were installed in a 120-passenger twin-engine short-range aircraft. Study results show the Prop-Fan powered aircraft has an advantage of 21% in fuel burned and 10% in direct operating cost over the comparable technology turbofan powered aircraft.

Many technology areas remain to be verified before industry can commit to the design and development of a Prop-Fan powered aircraft. The key engine-related technologies are: (1) the large horsepower size reduction gearbox, (2) Prop-Fan/nacelle/inlet/compressor interactions, and (3) the small size high-pressure compressor. Verification plans for these technologies are presented in Section 4.5.

Many more engine/aircraft integration studies remain to be conducted which may identify other technology needs. These are also discussed in Section 4.5 of this report.

The overall study followed the logical sequence of the tasks in the APET contract statement of work.

3.2 TASK I - SELECTION OF EVALUATION PROCEDURES AND ASSUMPTIONS

In Task I, Pratt & Whitney prepared a Procedures and Assumptions document to define the reference aircraft, aircraft mission, reference turbofan engine, direct operating cost methods and other key items in the study. The reference airplane and aircraft evaluation ground rules were developed with the assistance of four airframe companies: Boeing, Douglas, Lockheed-California and Lockheed-Georgia. The mission profile used for the design and typical missions follows U.S. rules with taxi times based on trunk airline experience. Air Transport Association domestic reserve requirements were used. The economic ground rules were based on the 1981 update of the Boeing direct operating cost procedure. After approval by the NASA Program Manager, the document served as the ground rules for the remainder of the study. The key procedures and assumptions are summarized in Table 3-I; a more detailed presentation is contained in Section 4.1 of this report.

The reference turbofan engine cycle, components, and configuration were selected based on work done in the benefit/cost portion of the NASA-sponsored Energy Efficient Engine program. Engine technology availability of 1988 and an engine certification date of 1992 were assumed to ensure compatibility with the Prop-Fan engine.

TABLE 3-I
KEY ASSUMPTIONS AND PROCEDURES

Engine Technology Availability	1988
Engine Certification	1992
Engine Sizing	2133 m (7000 ft) takeoff field length at sea level 29°C (84°F) day or 10,668 m (35,000 ft) initial cruise altitude capability
Reference Airplane	120-Passenger, Twin-Engine
Design Range	1800 Nautical Miles
Typical Flight Range	400 Nautical Miles
Cruise Conditions	0.75 Mach Number 10,668 m (35,000 ft) Altitude
Reference Turbofan Engine	Bypass Ratio = 7.0 Overall Pressure Ratio = 40.8 Combustor Exit Temp. = 1460°C (2660°F)
Reference Propeller	Hamilton Standard Ten Blade SP04A80
Unique Assumptions	
Cabin Acoustic Weight Penalty	Treatment Added to Reduce Cabin Noise to Turbofan Level
Propeller Slipstream Drag	0% (+ 3% Evaluated)

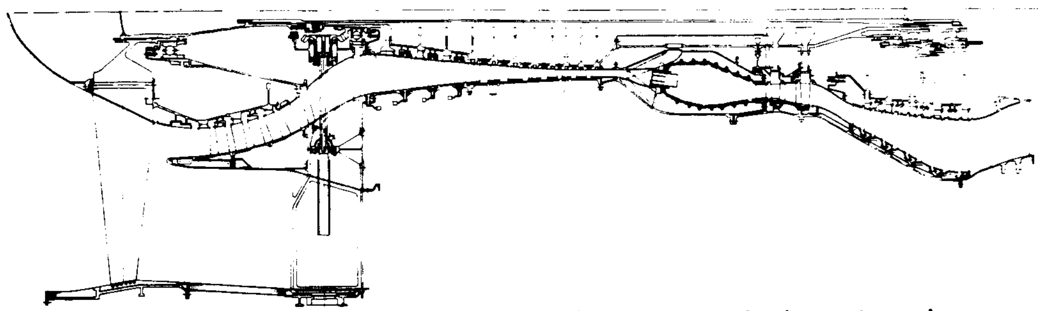
A cross section of the reference turbofan engine, designated STF686, is presented in Figure 3-1.

3.3 TASK II - CYCLE OPTIMIZATION AND ENGINE CONFIGURATION SELECTION

In Task II the Prop-Fan cycle was optimized using trade factors from Task I. These are shown in Table 3-II. The cycle optimization was done at three levels of horsepower to evaluate the effect on smaller and larger size engines.

Four different Prop-Fan engine configurations were evaluated. The two most promising candidates were a two-spool all-axial compression engine and a three-spool engine with axial/centrifugal compression.

The NASA Program Manager approved our selections of cycle and configurations and they were then used for the Task III Propulsion System Integration.



	Cycle description at cruise	
	STF686	E^3 technology
Thurst class, N (lb)	84 (19K)	177 (40K)
Overall pressure ratio	40.8	38.6
Bypass ratio	7.0	7.2
Fan pressure ratio	1.66	1.65
Maximum CET, C° (°F)	1460 (2660)	1437 (2620)
Exhaust type	Separate	Mixed

Figure 3-1 Reference Turbofan Engine (STF686) - The reference turbofan is based on work done in the benefit/cost portion of the Energy Efficient Engine program.

TABLE 3-II
PROP-FAN AIRPLANE TRADE FACTORS
(1981 Dollars, 0.396 per liter ($1.50 per Gallon))

Trade Factor	Effect on Fuel Burn	Effect on Direct Operating Cost
One percent increase in TSFC	1.12%	0.46%
10 pound increase in pod drag per engine	0.32%	0.15%
1000 pound increase in weight per engine	2.16%	1.29%
$100,000 increase in the price of each engine	---	0.27%
$10.00 increase in maintenance cost per engine flight hour	---	0.96%

3.3.1 Cycle Optimization

The cycle was optimized for a base engine rating of 16,000 shaft horsepower using the trade factors from Task I. Consideration was given to overall pressure ratios from 20 to 45 and maximum combustor exit temperatures from 1204 to 1537°C (2200 to 2800°F). The optimization process considered engine TSFC, propulsion system weight, mission fuel burned and direct operating cost as a function of overall pressure ratio and combustor exit temperature.

Figure 3-2 illustrates the TSFC, weight, fuel burned and DOC curves which were considered in selecting the cycle. A cycle of 35:1 design point overall pressure ratio and 1426°C (2600°F) maximum combustor exit temperature was selected considering both fuel burned and DOC. This cycle was used for the configuration selection portion of the study.

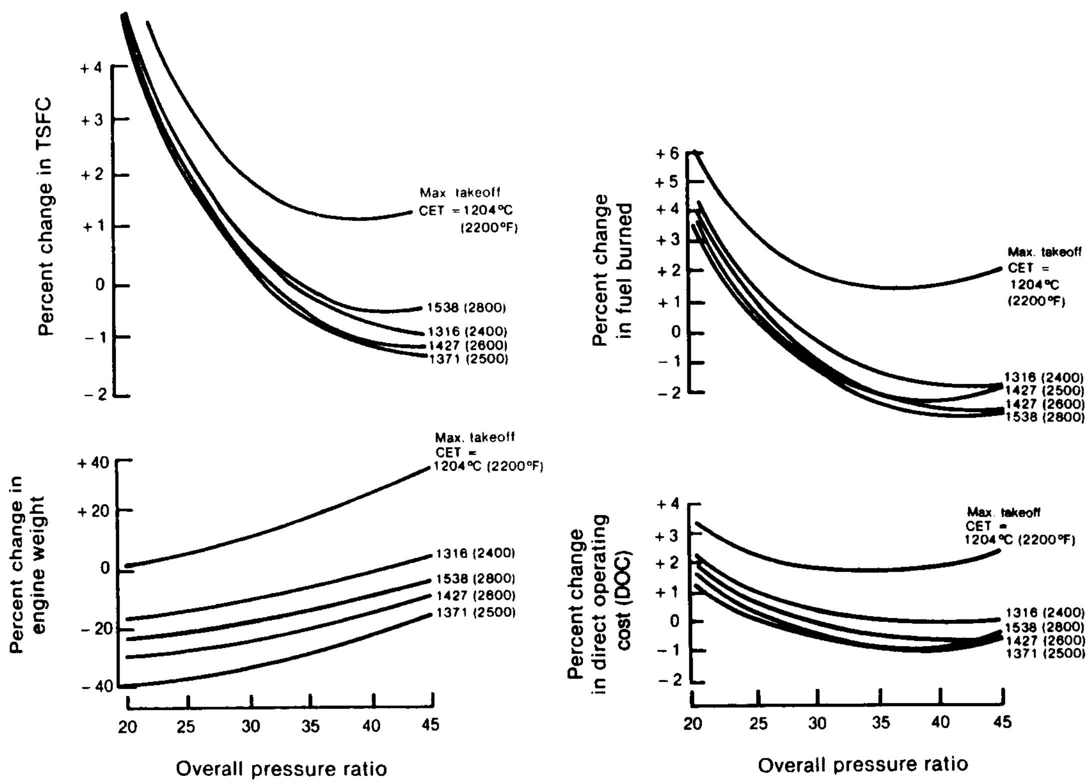


Figure 3-2 Factors Considered in Selecting the Optimum Cycle - These tradeoffs resulted in the selection of the 1,426°C (2600°F) CET, 35:1 OPR cycle for the 16,000 shaft horsepower engine.

Because of the uncertainty of the correct horsepower requirements the cycle optimization was also done at 8000 and 23,000 horsepower to properly reflect smaller and larger size engines. The overall pressure ratio and combustor exit temperature are fairly insensitive to variations in engine size between 8000 and 23,000 horsepower as indicated in Figure 3-3.

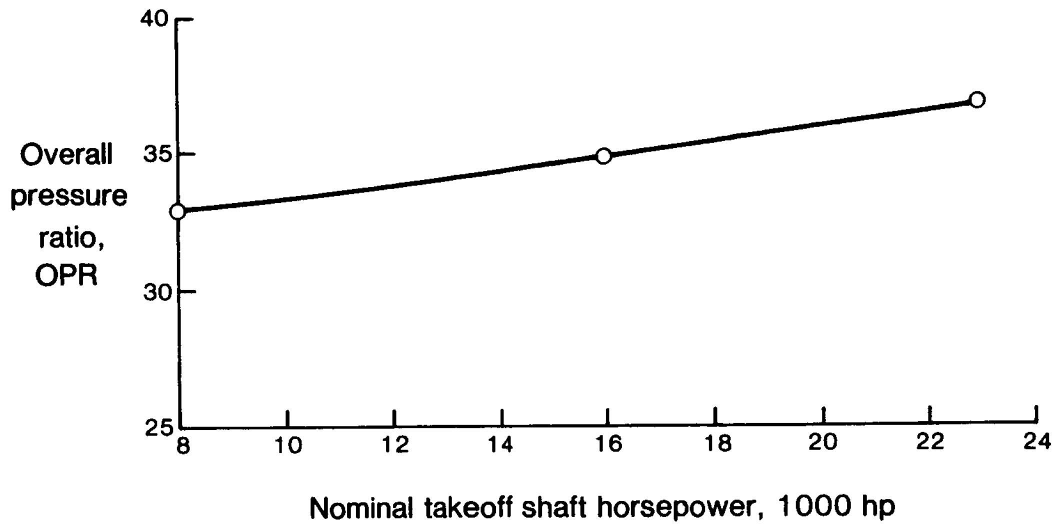
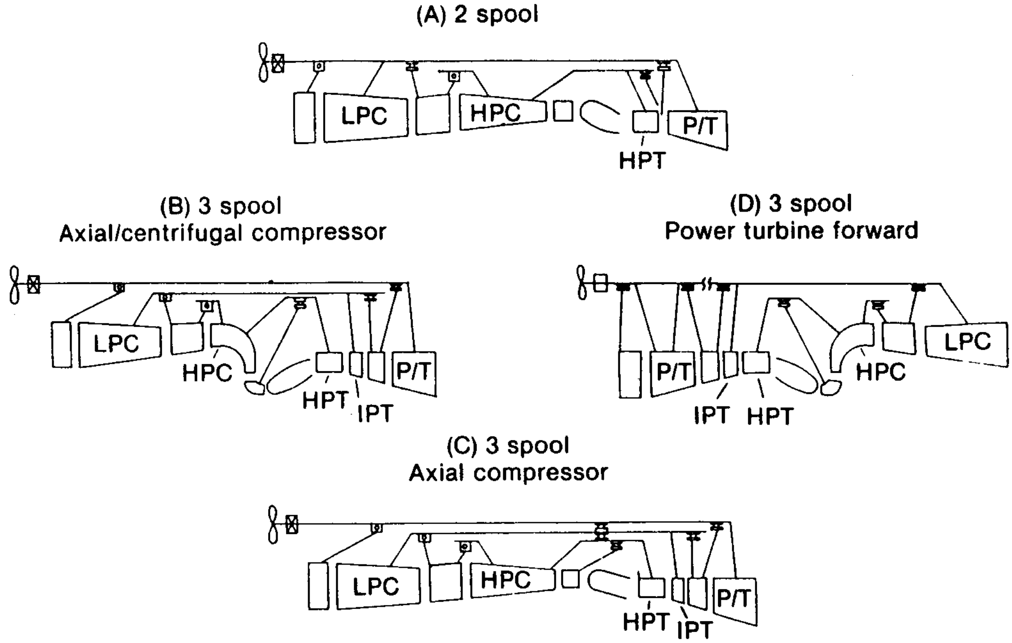


Figure 3-3 Overall Pressure Ratio as a Function of Engine Size at 1,426°C (2600°F) Maximum Takeoff Temperature - Overall pressure ratio is fairly insensitive to engine size.

3.3.2 Configuration Selection

Four different turboprop engine configurations were evaluated. They included: (A) a two-spool engine with all-axial compression, (B) a three-spool engine with axial/centrifugal compression, (C) a three-spool engine, like (B), but reversed so the inlet is at the rear and the turbine in the front, and (D) a three-spool engine with all-axial compression. The configurations and results of the evaluation are presented in Figure 3-4.

Of the four configurations evaluated, the two-spool engine with all-axial compression and the three-spool engine with axial/centrifugal compression demonstrated the best combination of fuel burned and direct operating costs; thus, these engine configurations were selected for further evaluation under Task III. The performance of the three-spool axial compression engine was inferior to the performance of the two-spool axial compression engine, while the direct operating cost of the three-spool axial compression engine was inferior to the operating cost of the three-spool axial/centrifugal compression engine. Poor fuel burn characteristics caused the reversed engine configuration to be eliminated from consideration.



Configuration	A	B	C	D
Fuel Burned	Base	+1.5%	+5.6%	+1.1%
Direct Operating Cost	Base	-0.9%	+0.8%	-0.1%

Figure 3-4 Turboprop Engine Configuration Candidates and Evaluation Summary - The all-axial two-spool and axial/centrifugal three-spool configurations were selected for further evaluation.

The two-spool axial compression engine and the three-spool axial/centrifugal compression engine are illustrated in Figures 3-5 and 3-6, respectively.

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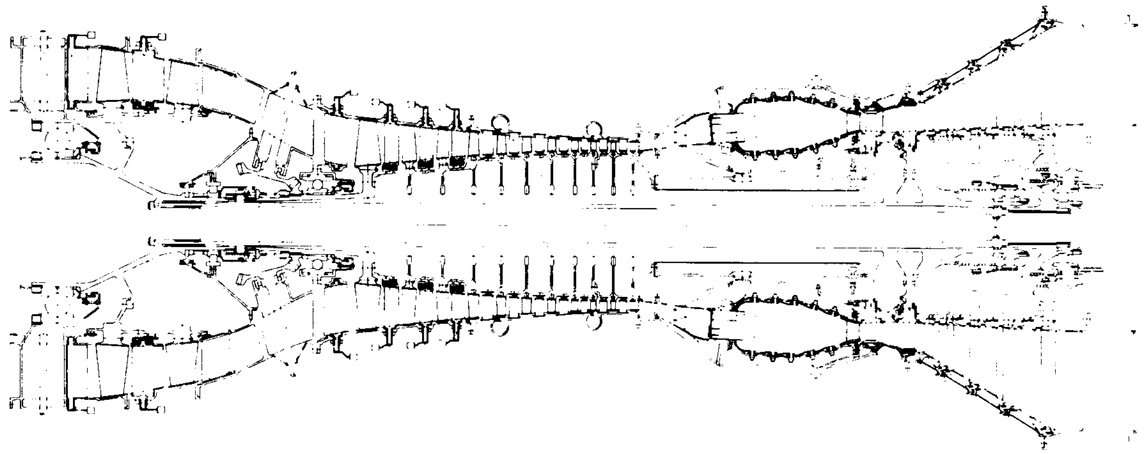

Figure 3-5 Two-Spool Engine With All-Axial Compression - This configuration was selected for further evaluation under Task III.

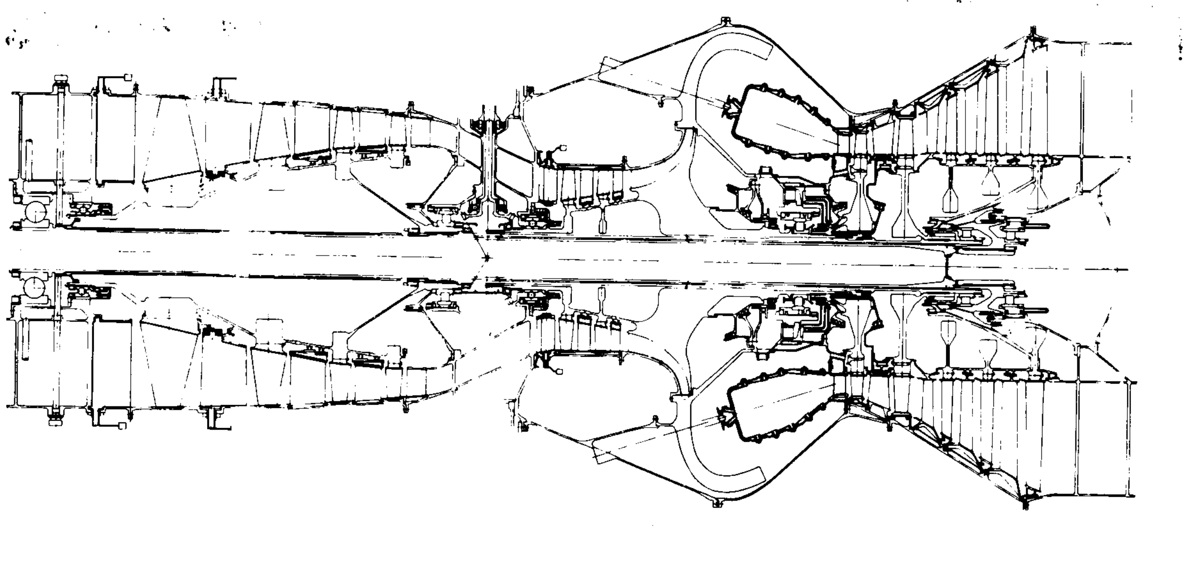

Figure 3-6 Three-Spool Engine With Axial/Centrifugal Compression - This configuration was selected for further evaluation under Task III.

3.4 TASK III - PROPULSION SYSTEM INTEGRATION

The two selected configurations were combined with the Prop-Fan, reduction gear, and nacelle in two integrated propulsion systems. These integrated propulsion system packages were given to the four airframe manufacturers for their evaluation. Input from the airframe manufacturers was considered when the NASA Program Manager and Pratt & Whitney mutually selected two propulsion system configurations for engine/aircraft evaluation in Task IV.

A computer deck simulating the performance of both propulsion systems was generated for the Task IV Engine/Aircraft Evaluation.

3.4.1 Selected Integrated Propulsion Systems

Using the two engine configurations selected in Task II, Pratt & Whitney prepared two integrated propulsion system packages for evaluation by the airframe companies and for further study under Task III Propulsion System Integration. These include:

- A two-spool engine, with a non-free power turbine drive, all-axial compression system, an offset compound idler reduction gear, a chin mounted inlet, and an oil-to-fuel cooling system with a supplemental oil-to-air cooler system for auxiliary use.

- A three-spool engine with a free turbine drive, an axial/centrifugal compression system, a split-path in-line reduction gear, a trifurcated duct inlet and an oil-to-air cooling system.

3.4.2 Airframe Companies' Review of Integration Package

A propulsion system integration package was sent to Boeing, Douglas, Lockheed-California and Lockheed-Georgia for their comments. This package is discussed in detail in Section 4.3. It included:

- Aircraft accessory options
- Engine/gearbox base size dimensions
- Configuration evaluation summary
- In-line/offset reduction gear options
- Gearbox oil cooler information
- Inlet configuration options and comparisons
- A system integration summary
- Conceptual nacelle for over-the-wing installation
- Propulsion system mounting options

Written replies were received from Lockheed-Georgia, Lockheed-California, and Boeing. These comments were integrated and are discussed in detail in Section 4.3.

3.4.3 Final Propulsion System Selections

Based on a consensus of the comments by the airframe companies, along with our own studies, two propulsion systems, presented in Figures 3-7 and 3-8, were selected for further evaluation under Task IV. These configurations were approved by the NASA Program Manager and are described in detail in Section 4.3 of this report. Some of the more pertinent selection details include:

- Both the in-line and offset reduction gear concepts will continue to be evaluated.
- The 12,000 shaft horsepower base engine size was chosen representing the power required for the reference aircraft and mission.
- An over-the-wing engine installation was selected based on input from the aircraft companies. Minimizing landing gear length was a significant factor in this selection.

3.4.4 Engine Performance Decks

Pratt & Whitney prepared a computer deck for calculating steady state performance for the 12,000 shaft horsepower base size engine. This deck represents the performance of both the two-spool and three-spool propulsion systems. Appropriate weight and dimensions are included for each of the two-spool and three-spool engines. This deck will also have provisions for scaling the engine performance, weight and dimensions over a range of 8000 to 23,000 shaft horsepower.

A computer deck for the comparable technology turbofan with appropriate scaling capability, was also provided under the contract.

User manuals for both decks were prepared and the entire package was submitted to NASA.

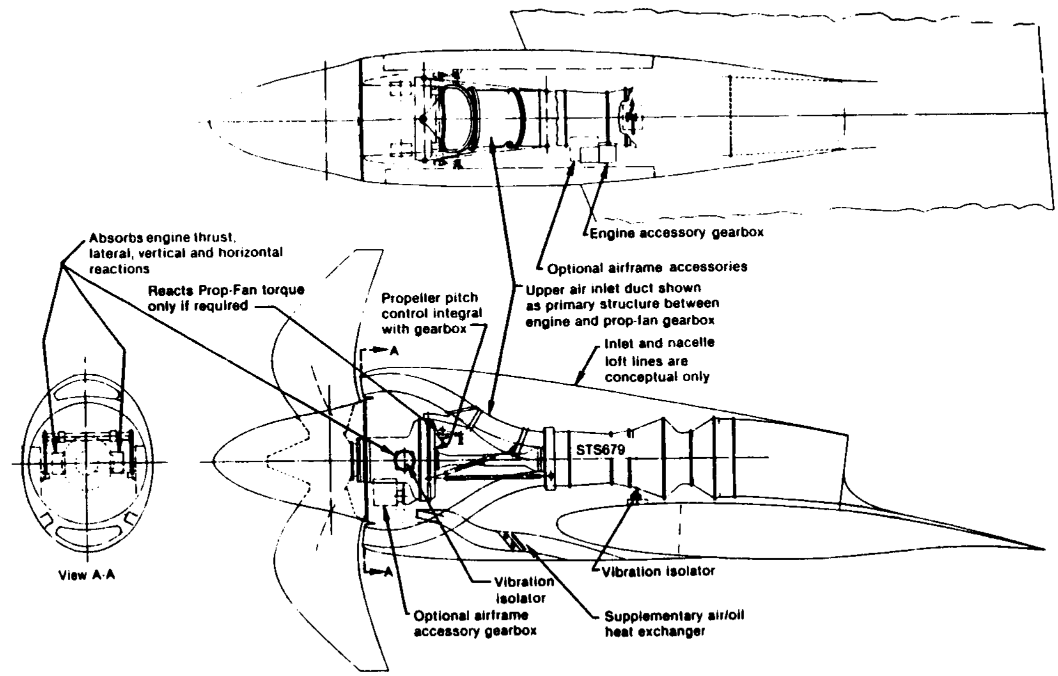

Figure 3-7 In-Line Gearbox Propulsion System Installation - This configuration was selected for further study under Task IV.

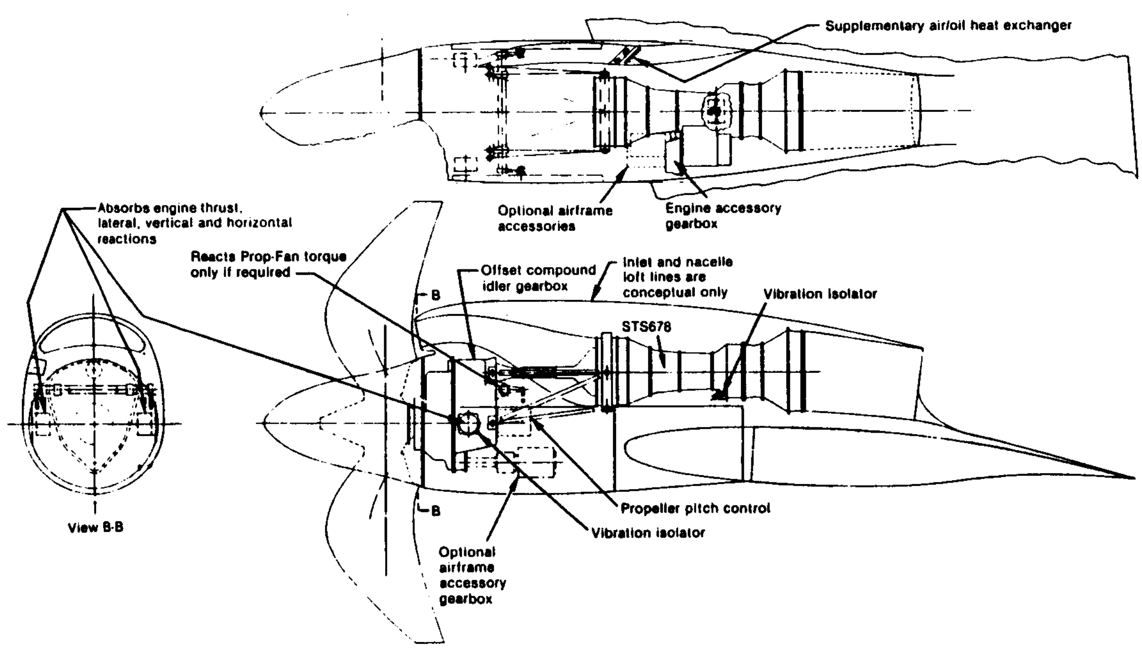

Figure 3-8 Offset Gearbox Propulsion System Installation - This configuration was selected for further study under Task IV.

3.5 TASK IV - ENGINE/AIRCRAFT EVALUATION

A mission simulation study was conducted using the Task I approved aircraft to assess the merits of the turboprop propulsion systems relative to a comparable technology turbofan engine. The aircraft were flown on the Task I reference mission and compared on the basis of fuel burned and direct operating costs. Mission studies indicated that the Prop-Fan powered aircraft has a potential 24% fuel burned and 12% direct operating cost advantage over a comparable technology turbofan powered aircraft for the typical mission.

3.5.1 Comparison of Turbofan and Prop-Fan Powered Aircraft

The engine characteristics for the mission study are compared in Table 3-III. Both the Prop-Fan and turbofan are advanced technology engines with 1992 certification. Both engines were installed in twin engine aircraft designed to carry 120 passengers on a 1800 nm mission and with a typical mission range of 400 nm at 0.75 Mach Number, 10,668 m (35,000 ft) cruise conditions.

TABLE 3-III
TURBOFAN AND PROP-FAN ENGINE CHARACTERISTIC COMPARISONS

	Turbofan	Prop-Fan
Bypass Ratio (BPR)	7.0	---
Overall Pressure Ratio (OPR) at Max Climb, 10,668 m (35,000 ft) Altitude	40.8	38.3
Combustor Exit Temperature (CET)		
Growth	2660	2600
Initial	2590	2530
Takeoff Power at Sea Level Standard Day Plus -3°C (25°F), Static Thrust, Mach 0.3 for shp	7,529 kg (16,600 lb)	11,600 shp
Engine Sizing Condition	Takeoff	Max Climb

A weight comparison of both aircraft is presented in Table 3-IV. The operating empty weight of the Prop-Fan powered aircraft is greater than that of the turbofan powered aircraft because the total propulsion system weight is higher and because there is a fuselage acoustic weight penalty for equal cabin noise levels. However, the takeoff gross weights are nearly equal due to the lesser amount of mission fuel carried by the more fuel efficient Prop-Fan powered aircraft.

TABLE 3-IV
WEIGHT COMPARISON OF PROP-FAN AND TURBOFAN POWERED AIRCRAFT
(0.75 Mn, 10,668 m (35,000 ft) Cruise Conditions)

	Weights kg (lb)	
	Prop-Fan	Turbofan
Takeoff Power at Sea Level Standard Day, +13°C (+25°F)	11,600 shp	73,840 N (16,600 lb)
Engine	810 (1787)	1383 (3051)
Reduction Gear	509 (1123)	---
Propeller	640 (1411)	---
Nacelle	781 (1723)	1116 (2461)
Total	2741 (6044)	2500 (5512)
Fuselage Acoustic Weight Penalty	935 (2062)	---
Aircraft Operating Empty Weight (OEW) lbs	33,833 (74,590)	32,418 (71,470)
Aircraft Takeoff Gross Weight (TOGW) lbs	52,480 (115,700)	52,661 (116,100)

3.5.2 Aircraft Mission Evaluation

Table 3-V presents the results of the comparison in terms of fuel burned and direct operating costs. Study results show that the Prop-Fan powered aircraft has significant advantages in fuel burned and direct operating costs (DOC) over the comparable technology turbofan powered aircraft. The study is discussed in detail in Section 4.4 of this report.

TABLE 3-V
COMPARISON OF PROP-FAN TO TURBOFAN POWERED AIRCRAFT
(0.75 Mn, 10,668 m (35,000 ft) Cruise Conditions)

	400 Nautical Miles (Typical Mission)	1800 Nautical Miles (Design Range)
Fuel Burned	-21%	-17%
Direct Operating Cost (DOC)	-10%	-8%

3.5.3 Environmental Considerations

Task IV of the APET program also addressed environmental considerations of the Prop-Fan propulsion system. It is expected that the Prop-Fan powered aircraft will meet all emissions and flyover acoustic regulations for 1992 certification. This is discussed in greater detail in Section 4.4.

3.5.3.1 Emissions

The emissions goals of the International Civil Aviation Organization were used in the APET study. These goals, presented in Table 3-VI, are referred to as "Research Goals" for newly certified engines. The advanced Mark V combustion system which is projected to be available for 1992 engine certification will provide the capability to meet these emissions goals for both the Prop-Fan and turbofan engines.

TABLE 3-VI
INTERNATIONAL CIVIL AVIATION ORGANIZATION
(Emissions Research Goals)

	Research Goals (g/kN)*	
	Prop-Fan	Turbofan
Unburned Hydrocarbons	4.35	4.35
Carbon Monoxide	42.0	42.0
Oxides of Nitrogen	54	56.6
Smoke (SAE Number)	24.7	24.4

* Thrust at Sea Level Takeoff Static Conditions in kilonewtons

3.5.3.2 Acoustics

The flyover noise of the Prop-Fan powered aircraft was estimated at the certification points defined by the FAA Part 36 Chapter 3 regulations. The Prop-Fan powered aircraft is predicted to meet FAR 36 noise regulations with margin at all three measuring points. Figure 3-9 presents these predictions for both the Prop-Fan and turbofan powered aircraft.

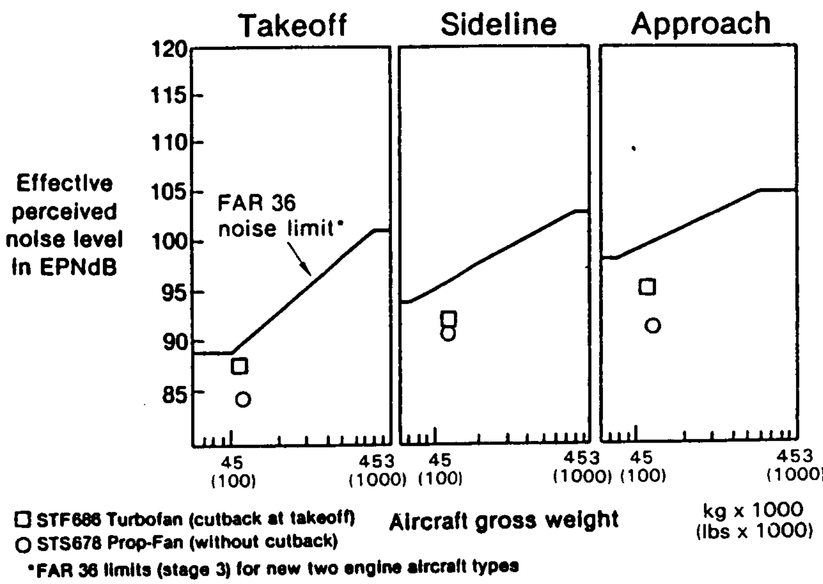

Figure 3-9 Noise Predictions - The Prop-Fan powered aircraft meets FAR 36 regulations with margin at all three measuring points. (J27638-149)

3.6 TASK V - ADVANCED PROP-FAN ENGINE TECHNOLOGY PLAN

The objectives of Task V were: (1) identify the key technology components for an advanced Prop-Fan engine system and (2) prepare a key engine technology development and verification plan.

3.6.1 Technology Identification and Component Verification Plan

Three component technology areas unique to the Prop-Fan propulsion system have been identified. They are:

- o Reduction gear/pitch control
- o Prop-Fan/nacelle/inlet/compressor interaction
- o Small-size high-pressure compressor

Detailed technology verification plans, including schedules and estimated costs for planning, for these key technology areas were prepared and are presented in Section 4.5.

3.6.2 Additional Engine/Aircraft Integration Studies

During Task III integration with the aircraft companies, many engine/aircraft issues arose which could not be resolved by Pratt & Whitney alone under the APET contract. A distinct possibility exists that these unresolved issues could result in more key propulsion system technology identification. We recommend that joint engine/aircraft studies be funded by NASA to resolve these issues which include:

- o Free power turbine versus non-free power turbine engine/aircraft integration study
- o Propulsion system engine/aircraft mounting study
- o Engine/aircraft heat rejection study
- o Integrated engine/aircraft control study

Details of these proposed studies can be found in Section 4.5.

3.7 COMPARISON WITH COUNTER ROTATION PROPULSION

Pratt & Whitney was involved in a 1982 study with Hamilton Standard and Lockheed-Georgia to evaluate a counter rotation Prop-Fan and compare the results to single rotation Prop-Fan propulsion. Lockheed-Georgia results indicate the counter rotation system has the potential for 8% fuel burned and 2.5% direct operating cost improvement over the single rotation system.

These results require a word of caution. There is no model test background for the performance and acoustic predictions for the counter rotation propellers. Model tests must be run to put the counter rotation system on the same technical base as the single rotation propeller. If it is meaningful to conduct these tests, then they should be funded separately from the single rotation program to prevent dilution of the presently planned NASA program.

Wind tunnel tests should also be conducted on a supercritical wing installation to compare with the present Ames and Langley single rotation tests. These NASA tests indicate that the wing is a good straightener of the single rotation swirl flow. This in turn suggests that the benefit for the counter rotation propeller may be much less than the projected 8% from the joint study.

3.8 RELEVANCY OF PROP-FAN PROPULSION TO MILITARY APPLICATIONS

The excellent fuel economy of the Prop-Fan offers several opportunities for Military applications. These could include use in tactical transports and cargo planes or use in anti-submarine aircraft.

Studies are now being conducted by all three major aircraft manufacturers concerning C130 replacement aircraft. The Prop-Fan may be a natural for this application with its good fuel economy and excellent takeoff and reverse power characteristics which permit small field operation.

Previous Air Force studies have shown large advantages for a Prop-Fan powered aircraft in fuel savings, longer range, greater lifting capability and lower life cycle cost for cargo applications like the C-141.

The greater fuel efficiency of the Prop-Fan could permit application to anti-submarine warfare aircraft to permit aircraft to stay on station much longer.

Finally, the APET reduction gear program will provide excellent technology transfer to the Army helicopter reduction gear efforts along with future military turboprop applications.

For the Prop-Fan propulsion system to be effective in military applications, a program should be initiated to reduce the radar cross section effect of the Prop-Fan blading.

SECTION 4.0
DISCUSSION OF RESULTS

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Section 4.1 -- DISCUSSION OF RESULTS
Task I -- Selection of Evaluation Procedures and Assumptions

4.1 TASK I - SELECTION OF EVALUATION PROCEDURES AND ASSUMPTIONS

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4.1 TASK I - SELECTION OF EVALUATION PROCEDURES AND ASSUMPTIONS

4.1.1 Introduction

The objective of Task I was to establish comprehensive procedures and assumptions for the APET Definition Study. Key issues which were addressed include:

- o Engine technology availability and certification timing
- o Reference aircraft and mission
- o Reference turbofan
- o Turboprop propulsion system
- o Engine/airplane trade factors
- o Economic considerations and environmental constraints

Preliminary ground rules were developed using results from previous Pratt & Whitney studies. These preliminary ground rules were then reviewed by the four airframe manufacturers participating in the APET study (Boeing, McDonnell Douglas, Lockheed-Georgia, and Lockheed-California) and written critiques were submitted to Pratt & Whitney. Working with the NASA Program Manager, Pratt & Whitney modified the study ground rules wherever feasible to reflect the comments and suggestions of the airframe manufacturers. After the Study Procedures and Assumptions Document (Reference 1) was approved by NASA in March 1982, work began on the remaining technical tasks.

This section summarizes key facets of the procedures and assumptions used in the APET study.

It should be noted that Task I specified the minimum level of effort required to achieve the objectives of the APET Program. In many cases, the technical effort was expanded significantly beyond these minimum levels.

4.1.2 Engine Technology Availability and Certification Timing

Market studies indicated that the most promising application for Prop-Fan propulsion is the short-to-medium range, 100-120 passenger, replacement aircraft market, starting in the early 1990's. In order to certify a Prop-Fan powered aircraft by 1992, key propulsion system technologies have to be verified by 1988. Both the advanced turboprop propulsion systems and the reference turbofan engine evaluated in the APET study incorporate technology features and cycle parameters appropriate for 1988 technology verification and 1992 commercial engine certification.

4.1.3 Reference Aircraft and Mission

A 120-passenger commercial transport was selected as the reference aircraft for the APET Definition Study. A mission consisting of a 3333 km (1800 nm) design range and a 740 km (400 nm) typical mission was considered representative for airplanes in this class.

4.1.3.1 Reference Aircraft

The key features of the 120-passenger reference aircraft are summarized in Table 4.1-I. An airplane of this size was considered well-suited to the commercial market of the 1990's for several reasons. First, current 100 to 140 passenger aircraft, first purchased in the 1960's, will be candidates for replacement by 1990. Second, new airplanes have been proposed, and in some cases built, for the 150-220 passenger market. These airplanes will be in their prime in the 1990's and thus will not be candidates for replacement. Finally, the trend to hub-spoke airline route systems and the abandonment of service to smaller cities by trunk carriers will relegate airplanes with less than 100 seats to local routes where the speed capability of the Prop-Fan will not be required.

A cruise Mach number of 0.75 was selected as representative of the cruise speeds for airplanes in the 120 passenger class.

The cabin acoustic weight penalty required to achieve levels of cabin noise comparable to a turbofan powered aircraft (82 dB) was calculated to be approximately 1.7% of the takeoff gross weight of the Prop-Fan powered airplanes. A propeller slipstream interference drag of zero (equal to turbofan interference) was assumed on the recommendation of the NASA Program Manager. However, the effect of a higher drag penalty (3%) was also evaluated.

TABLE 4.1-I
REFERENCE AIRCRAFT DESCRIPTION

<u>Reference Aircraft</u>

Type:	Commercial passenger transport with two wing-mounted engines, Mach 0.75 cruise.
Size:	120 passengers in 10/90 first/tourist class split, 96 cm (38 in)/86 cm (34 in) seat pitch, six abreast, single aisle.
Technology:	Pratt & Whitney projection for airplanes entering service in the early 1990's.
Interior Noise:	Cabin interior noise goal was 82 dBA.

<u>Reference Missions</u>

Design:	3333 km (1800 nm) with 120 passengers, no cargo, U.S. rules, ATA domestic reserves, Mach 0.75 cruise.
Typical:	740 km (400 nm) with 72 passengers, no cargo, U.S. rules, ATA domestic reserves, Mach 0.75 cruise.

TABLE 4.1-I (Continued)
REFERENCE AIRCRAFT DESCRIPTION

Engine Sizing Conditions

The more critical of:	Takeoff field length (FAR) of 2133 m (7000 ft) at sea level, 28°C (84°F), or Initial cruise altitude capability on design mission of 10,668 m (35,000 ft) (the effect of 9448 m (31,000 ft) was also examined).

Reference Prop-Fan

Type:	Hamilton Standard Division 10 blade model from data package SP04A80, October 1980.
Tip Speed:	243 m/sec (800 ft/sec)
Disk Loading:	34.1 shp/D^2 at 10,668 m (35,000 ft), M 0.75, maximum climb. The 34.1 power loading permits 10% growth without change in Prop-Fan diameter.

4.1.3.2 Mission Profile

The missions chosen for this study followed directly from the choice of airplane size. A 3333 km (1800 nm) design range and a 740 km (400 nm) typical mission were considered representative of airplanes in the 120 passenger class. The 60 % load factor also follows current experience.

The mission profile used for the design and typical missions (Figure 4.1-1) followed U.S. rules, with taxi time based on trunk line experience. Air Transport Association domestic reserves were used. Cruises were flown at optimum altitude, subject to 1220 m (4000 ft) steps (9450, 10,670, 11,890 m (31,000, 35,000, 39,000 ft)) and thrust limitations.

4.1.4 Reference Turbofan

The reference turbofan selected for the APET study, designated the STF686, is a 8452 kN (19,000 lb) takeoff thrust, high bypass ratio engine incorporating technology features and cycle parameters appropriate for commercial engine certification in the 1992 time period. The Maximum Efficiency Energy Efficient Engine configuration identified in the Energy Efficient Engine program (NASA Contract NAS3-20646) provided the basis for the STF686 engine. The STF686 engine incorporates technology features four years beyond those incorporated in the Maximum Efficiency Energy Efficient Engine (scheduled for certification in 1988) which would improve thrust specific fuel consumption (TSFC) by 2 to 3% over the flight regime. However, the STF686 is smaller and does not include a mixer.

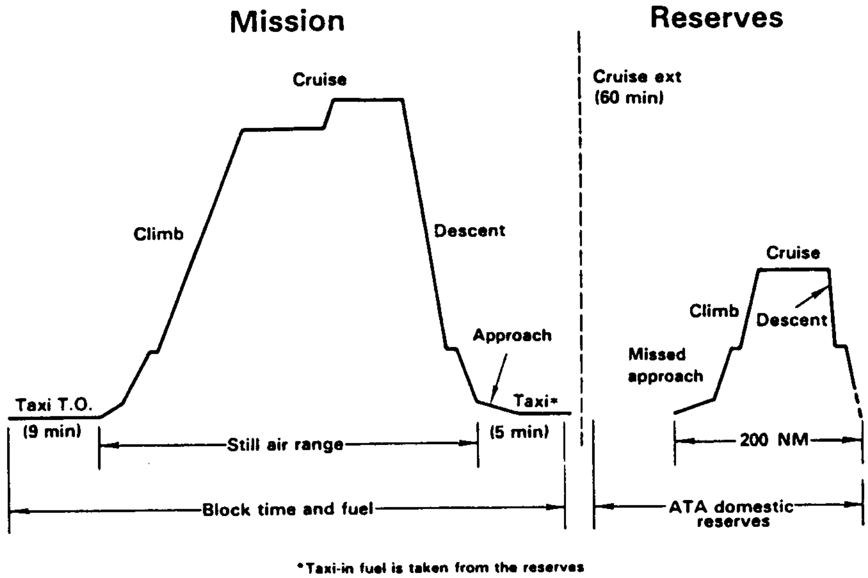

Figure 4.1-1 Nominal Mission Profile - The 3333 km (1800 nm) design range and 740 km (400 nm) typical mission are considered representative for 120 passenger aircraft. (J27638-901)

The STF686 has been configured as a separate flow engine. Discussions with aircraft manufacturers indicated that this configuration is consistent with use in short range aircraft applications.

Table 4.1-II presents a cycle description of the STF686.

TABLE 4.1-II
STF686 CYCLE DESCRIPTION
(0.75 Mn, 10,668 m (35,000 ft) Cruise Conditions)

Maximum Overall Pressure Ratio	40.8
Maximum Combustor Exit Temperature - °C (°F)	1460 (2660)
Fan Pressure Ratio	1.66
Bypass Ratio	7.0

A conceptual cross section drawing of the STF686 is presented in Figure 4.1-2. The high-pressure spool consists of an eleven-stage high-pressure compressor (pressure ratio 17:1) driven by a two-stage high-pressure turbine and a MARK combustion system. The low-pressure spool consists of a single-stage shroudless fan and a four-stage low-pressure compressor driven by a five-stage low-pressure turbine.

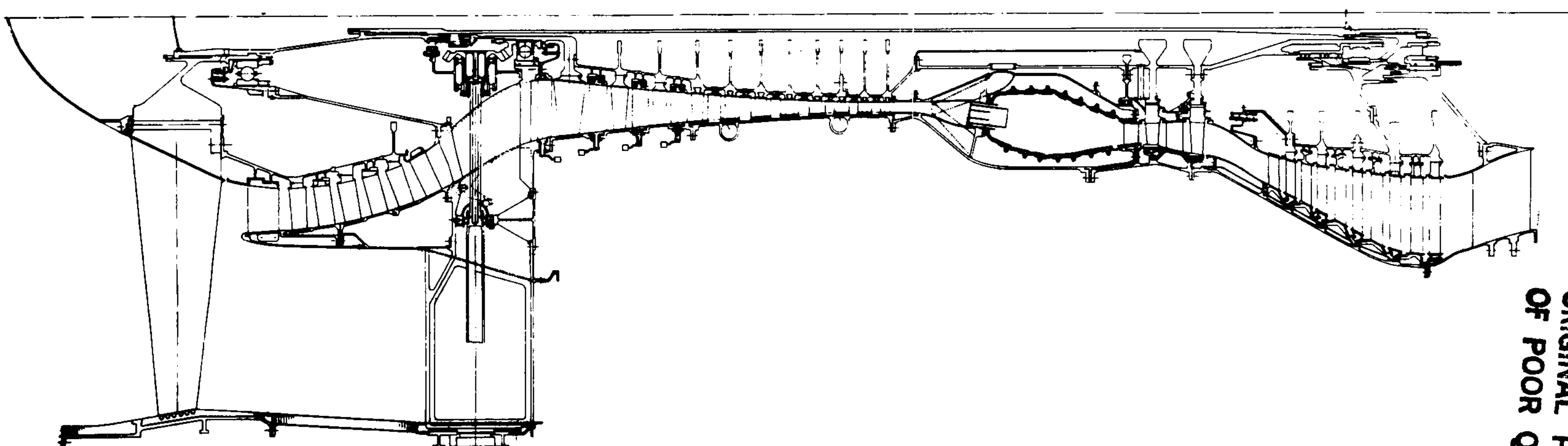

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Figure 4.1-2 STF686 Cross Section (Conceptual) - The STS686 turbofan engine incorporates technology features appropriate for 1992 commercial certification. (J27638-902)

Figure 4.1-3 contains performance data for the STF686 at altitudes of 9144 m (30,000 ft) and 10,668 m (35,000 ft) for flight Mach numbers from 0.6 to 0.8. The figure does not include nacelle drag which is charged against airplane performance. Takeoff performance is shown in Figure 4.1-4.

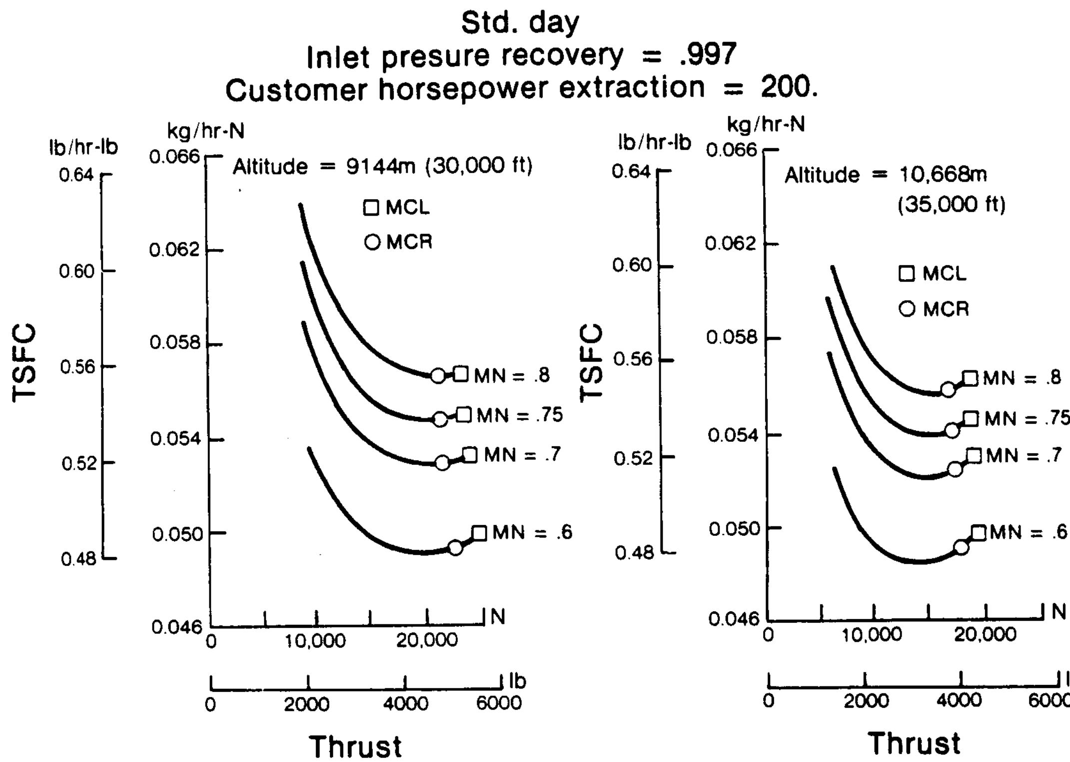

Figure 4.1-3 STF686 Performance at Altitude - Performance data is presented for cruise altitudes of 9,144 m (30,000 ft) and 10,668 m (35,000 ft). (J27638-213)

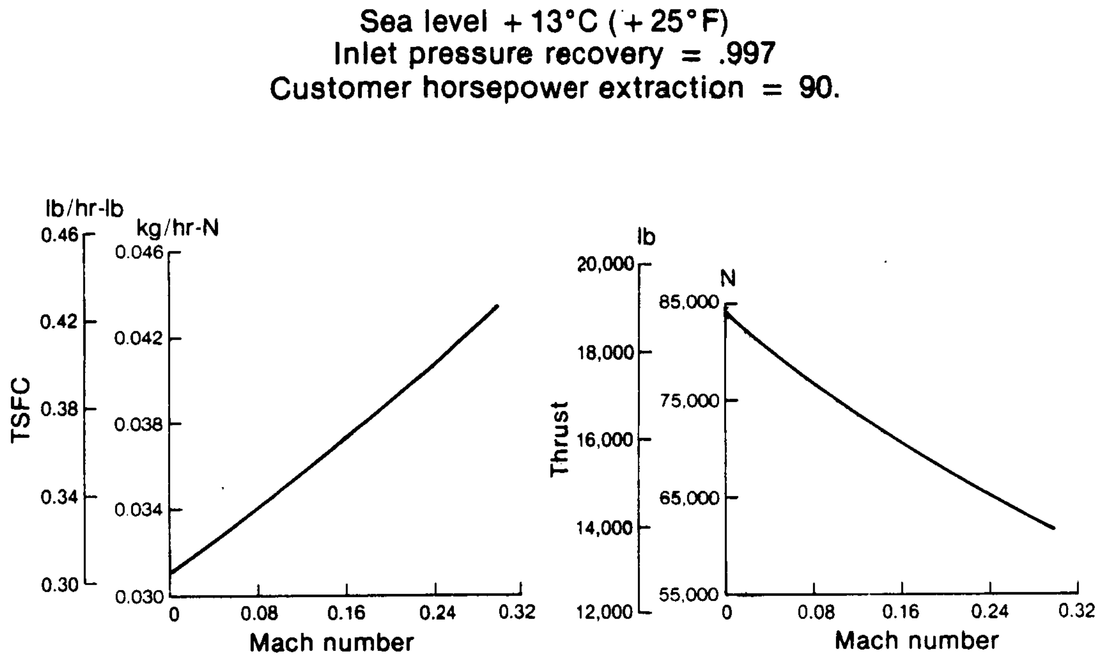

Figure 4.1-4 STF686 Takeoff Performance - Data is presented in terms of TSFC as a function of Mach number. (J27638-212)

Engine Components

The STF686 engine components incorporate technology advances expected to be available for 1992 certification.

Fan - The STF686 incorporates a single stage, 2.8 aspect ratio shroudless fan with increased flow capacity and higher aerodynamic loading. An improved airfoil contour will reduce shock losses and the manufacture of the airfoil contour with closer tolerances and consistency will improve fan performance.

Compressors - The low- and high-pressure compressors incorporate aerodynamic improvements including new airfoil contours and reduced endwall losses. Advances in airfoil contour design will come from better understanding of both the two-dimensional and three-dimensional loss mechanisms. The introduction of controlled diffusion airfoils (CDA's) in the early 1980's will be followed by a second generation of CDA's in the late 1980's. Improved three-dimensional modeling of endwall flow interactions will result in airfoil designs that enhance aerodynamic efficiency. Also, improvements in materials and mechanical configurations will allow better tip clearance management with active clearance control and new stator cavity designs resulting in improved compressor performance.

Combustor - The STF686 incorporates an advanced technology MARK V combustion system that is now under evaluation and development at Pratt & Whitney. It is an outgrowth of the combustor concepts developed under the NASA/Pratt & Whitney Experimental Clean Combustor Program and the NASA/Pratt & Whitney Energy Efficient Engine Program.

The MARK V combustion system uses high mixing rate technology to produce rapid burning and combustion product dilution with an integrated low pressure loss diffuser system.

Turbines - The major technology features in the turbine are improved single crystal airfoil materials and increased cooling effectiveness. These advances result in increased high pressure turbine efficiency and reduced turbine cooling requirements.

Improved single crystal airfoil materials permit higher stress turbine blade root designs (increased AN^2). This will in turn permit a better selection of aerodynamic parameters (load factors, H/U^2, and axial velocity ratio, Cx/U) for improved performance.

Improved single crystal airfoil materials, addition of thermal barrier coating on the blades and vanes, and increased cooling effectiveness will result in lower cooling airflow requirements and higher allowable compressor discharge temperature. Greater cooling effectiveness is attained by multipass designs which use impingement leading and trailing edges. Leading edge impingement air is reused as film through showerhead holes and trailing edge impingement air is used for convective cooling through the trailing edge holes. Skewed trip strips provide heat transfer augmentation. Film cooling is provided in the blade trailing edge tip regions.

Weight Estimate

The engine weight is 1587 kg (3500 lb).

Installation

The engine installation drawing is shown in Figure 4.1-5.

Acoustic Liners

The reference turbofan will satisfy Government noise regulations by incorporating acoustic liners in the engine and nacelle. Either perforated plate over honeycomb or wire mesh over perforated plate over honeycomb liner designs will be used. Locations at which the liners could be installed are shown in Figure 4.1-6. The liners in the fan inlet and discharge ducts and the fan case will be tuned to provide maximum attenuation of fan-generated noise. The liners in the primary nozzle will be tuned to attenuate turbine noise.

Liner designs will be based on the latest technology available. The location and amount of liner material will ensure that the airplane meets noise limits set forth in the Federal Aviation Regulations, Part 36, Stage 3.

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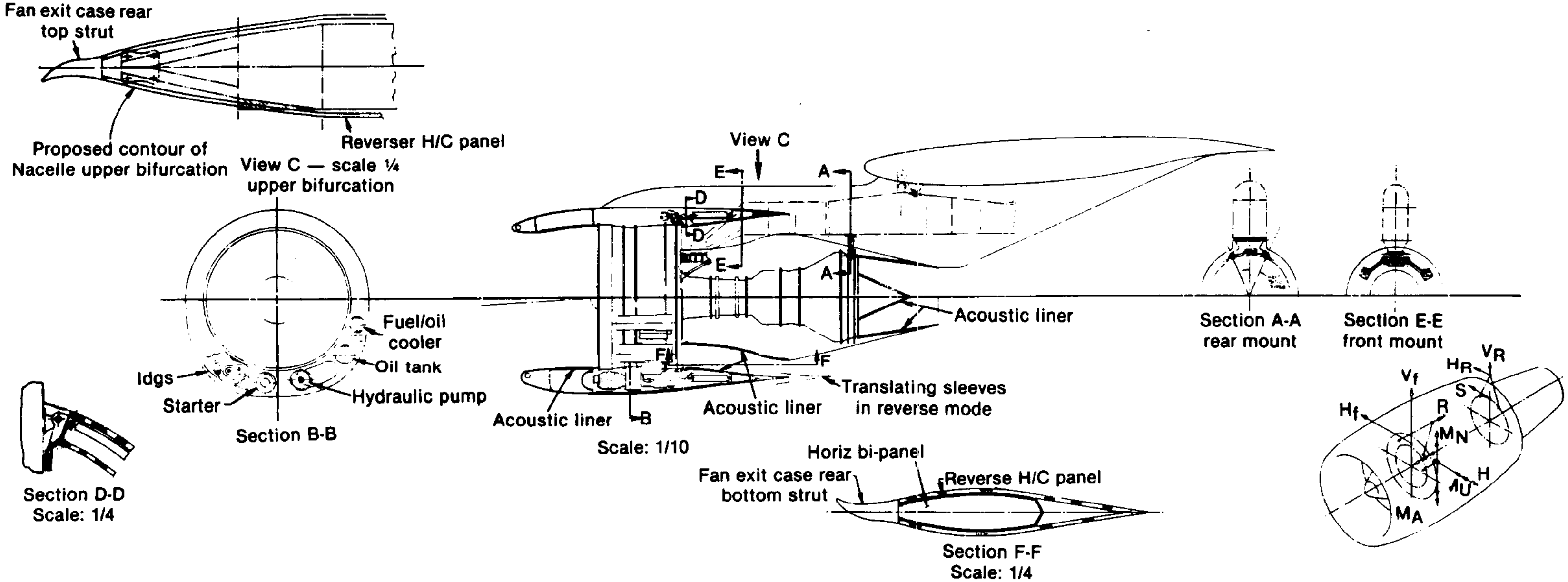

Figure 4.1-5 STF686 Installation Drawing - A typical turbofan pylon installation has been selected. (J27638-903)

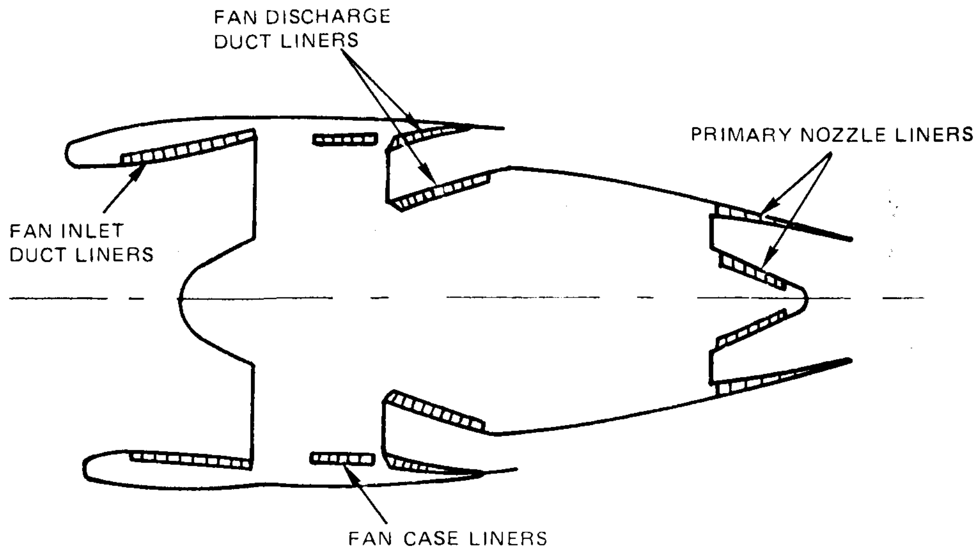

Figure 4.1-6 Acoustic Liner Locations (Schematic) - With acoustic treatment, the STF686 will satisfy noise limits specified in Federal Aviation Regulations. (J27638-904)

4.1.5 Turboprop Propulsion System

Four candidate turboprop engine configurations were evaluated in the APET Program. Prop-Fan performance characteristics were examined under a variety of conditions to aid in defining an optimum propeller system. In addition, many promising concepts were studied for three components unique to the turboprop propulsion system: reduction gear, heat rejection system, and inlet.

4.1.5.1 Study Turboprop Engine Configurations

Four candidate turboprop engine configurations (shown schematically in Figure 4.1-7) were selected to provide a variety of approaches to turboprop propulsion. They encompassed the use of two and three spools, axial and axial/ centrifugal compressors, and free and non-free power turbines. This variety of candidates provided a great deal of flexibility in conducting the Engine Configuration and Cycle Evaluation (Task II of the APET Program).

The two-spool configuration (Figure 4.1-7A) was used to explore the potential of using a turbofan-type high spool for a turboshaft application. The power turbine drives both the low-pressure compressor and the Prop-Fan in this configuration.

The two three-spool configurations (Figures 4.1-7B and 4.1-7C) permitted evaluation of a free power turbine relative to the two-spool non-free turbine configuration. These two configurations also permitted evaluation of the relative merits of axial versus axial/centrifugal compressors.

In the novel three-spool approach (Figure 4.1-7D), the inlet and compressor are at the rear and the turbines in front, directly behind the Prop-Fan. This arrangement of components permitted evaluation of unconventional aerodynamic and mechanical installation concepts. Several possible benefits relative to conventional systems were weighed against the penalties imposed by the unconventional arrangements. Among the potential benefits are simplified Prop-Fan/inlet integration because the inlet is aerodynamically remote from the Prop-Fan flow field. In addition, the two-spool engine could be used with a free third spool without requiring a third concentric shaft. These benefits were compared to the inlet and exhaust ducting losses associated with high flow turning angles.

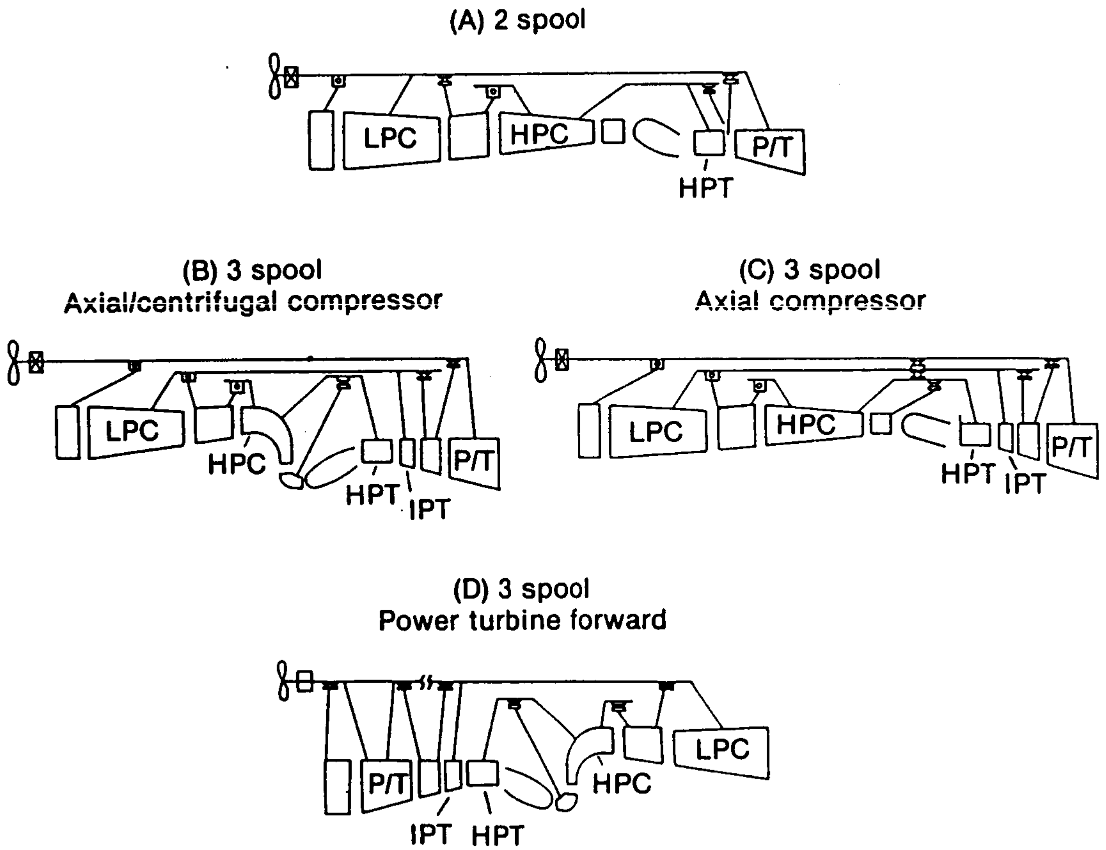

Figure 4.1-7 Candidate Turboprop Engine Configurations - The four candidate configurations cover a variety of approaches to turboprop propulsion. (J27638-159)

4.1.5.2 Prop-Fan Performance Characteristics

The effects of tip speed and disk loading (shp/D^2) variations on the efficiency (ETAPROP) of the Prop-Fan are shown in Figure 4.1-8. Disk loadings are quoted at maximum climb power setting, 10,668 m (35,000 ft) altitude, Mach 0.75. Combining the Prop-Fan with an advanced technology turboshaft engine produces the thrust and thrust specific fuel consumption characteristics shown in Figure 4.1-9.

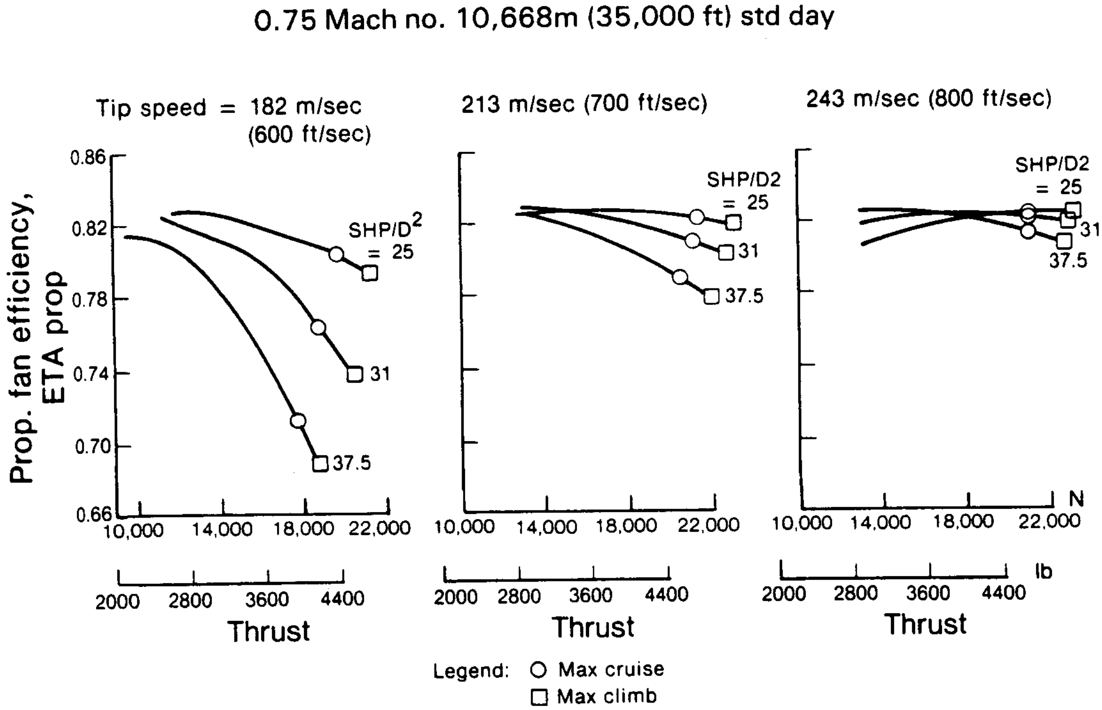

Figure 4.1-8 Sensitivity of Prop-Fan Efficiency to Tip Speed and Power Loading - A tip speed of 243 m/sec (800 ft/sec) ensures efficient operation at a variety of disk loadings. (J27638-129)

0.75 Mach no. 10,668m (35,000 ft) std day

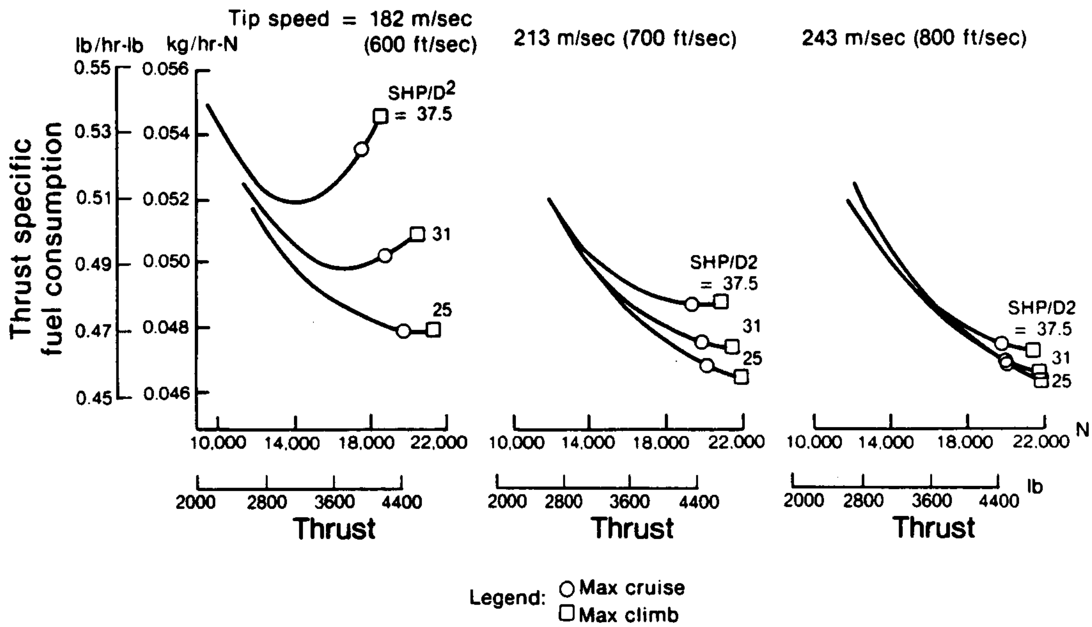

Figure 4.1-9 Sensitivity of TSFC to Prop-Fan Tip Speed and Power Loading - A tip speed of 243 m/sec (800 ft/sec) and disk loading between 31 and 37.5 shp/D^2 provide excellent thrust specific fuel consumption at both climb and cruise. (J27638-130)

4.1.5.3 Unique Components for a Turboprop Propulsion System

The reduction gear, heat rejection system, and inlet are unique components for a Prop-Fan propulsion system. The most promising concepts identified in previous studies were evaluated in the APET Program.

Reduction Gear Configurations

Based on Pratt & Whitney studies and input from the four airframe manufacturers, two reduction gear configurations were selected. The first is an offset compound idler configuration, shown in Figure 4.1-10. The second configuration, an in-line reduction system (see Figure 4.1-11), features an in-line split path concept. The overall efficiency for the reduction gear systems will be 99% at cruise.

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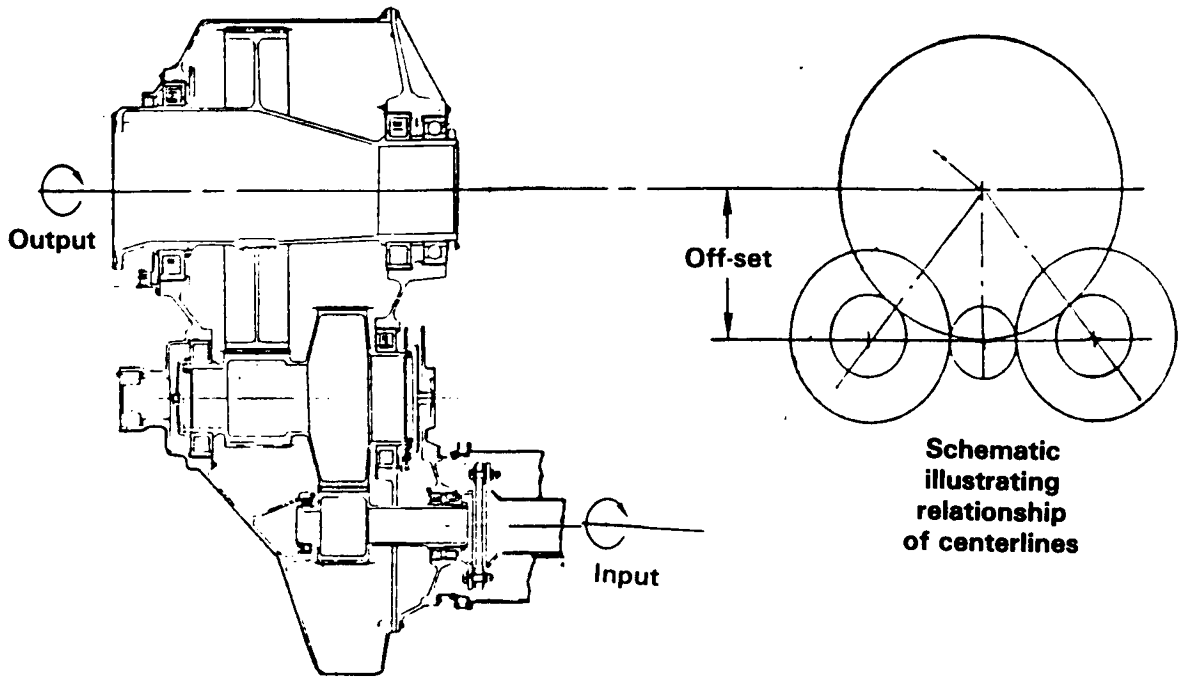

Figure 4.1-10 Compound Idler Offset Reduction Gear - This system features a minimum number of gears and bearings. (J27638-905)

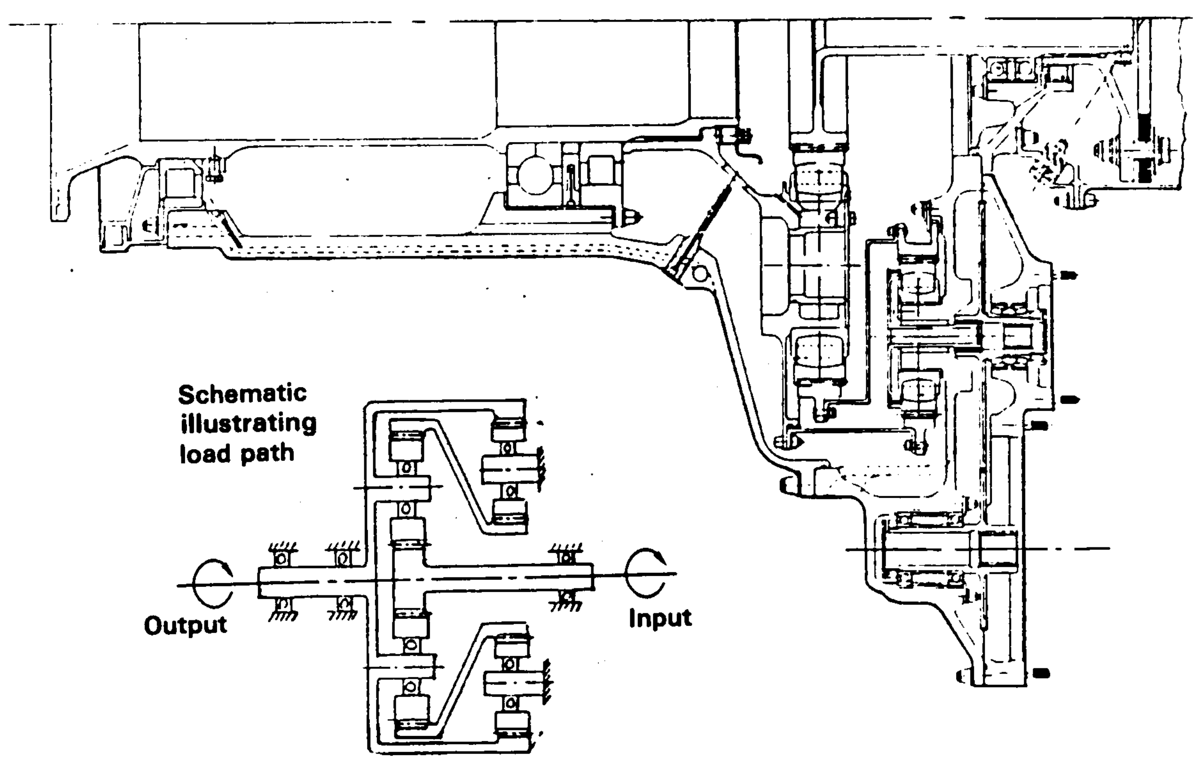

Figure 4.1-11 Split Path In-Line Reduction Gear - This system features minimum diameter and weight. (J27638-906)

Heat Rejection System Concepts

Two heat rejection systems were selected for evaluation: the double-flap inlet air/oil cooler concept (Figure 4.1-12) and a fuel/oil cooler system using aircraft fuel as a heat sink. The systems were sized to dissipate heat in the oil generated by the reduction gear. The critical conditions for sizing the heat exchanger are takeoff, maximum power, and ground idle.

The selection of two promising concepts was based on a comprehensive evaluation covering a variety of candidates. This evaluation is described in Section 4.3.

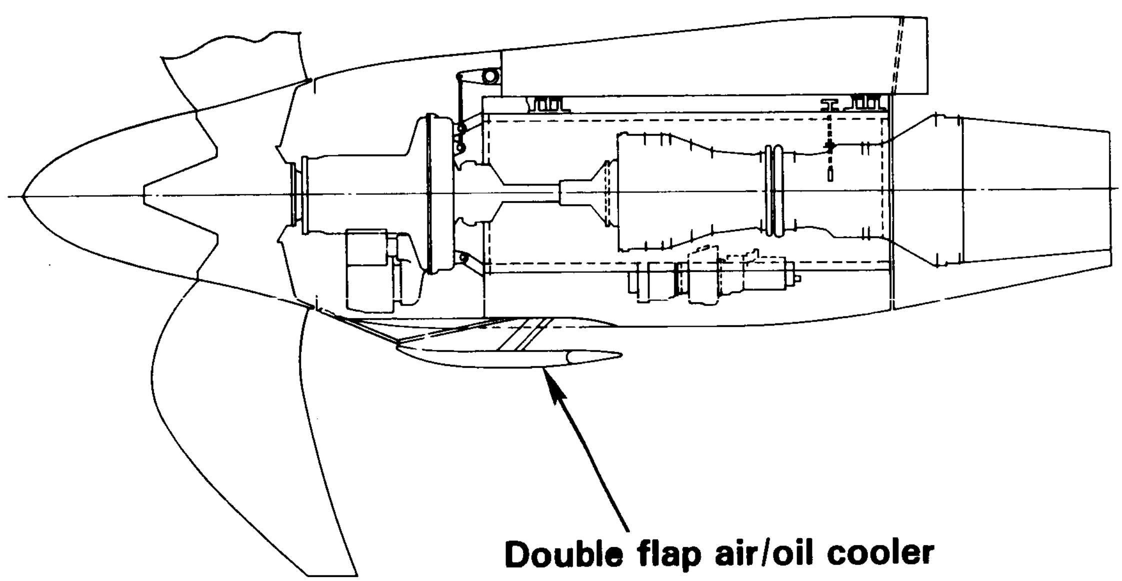

Figure 4.1-12 Double Flap Air/Oil Cooler - This system minimizes drag at the cruise operating condition. (J27638-907)

Inlets

Annular, bifurcated, and chin inlets were selected for evaluation in the APET study (see Figure 4.1-13). The chin inlet is primarily compatible with an offset gearbox while the annular inlet is most suited to an in-line gearbox. The bifurcated inlet is shown with an in-line gearbox, but could be adapted to an offset system.

The selection of three promising inlet concepts was based on a comprehensive evaluation covering a variety of candidates. This evaluation is described in Section 4.3.

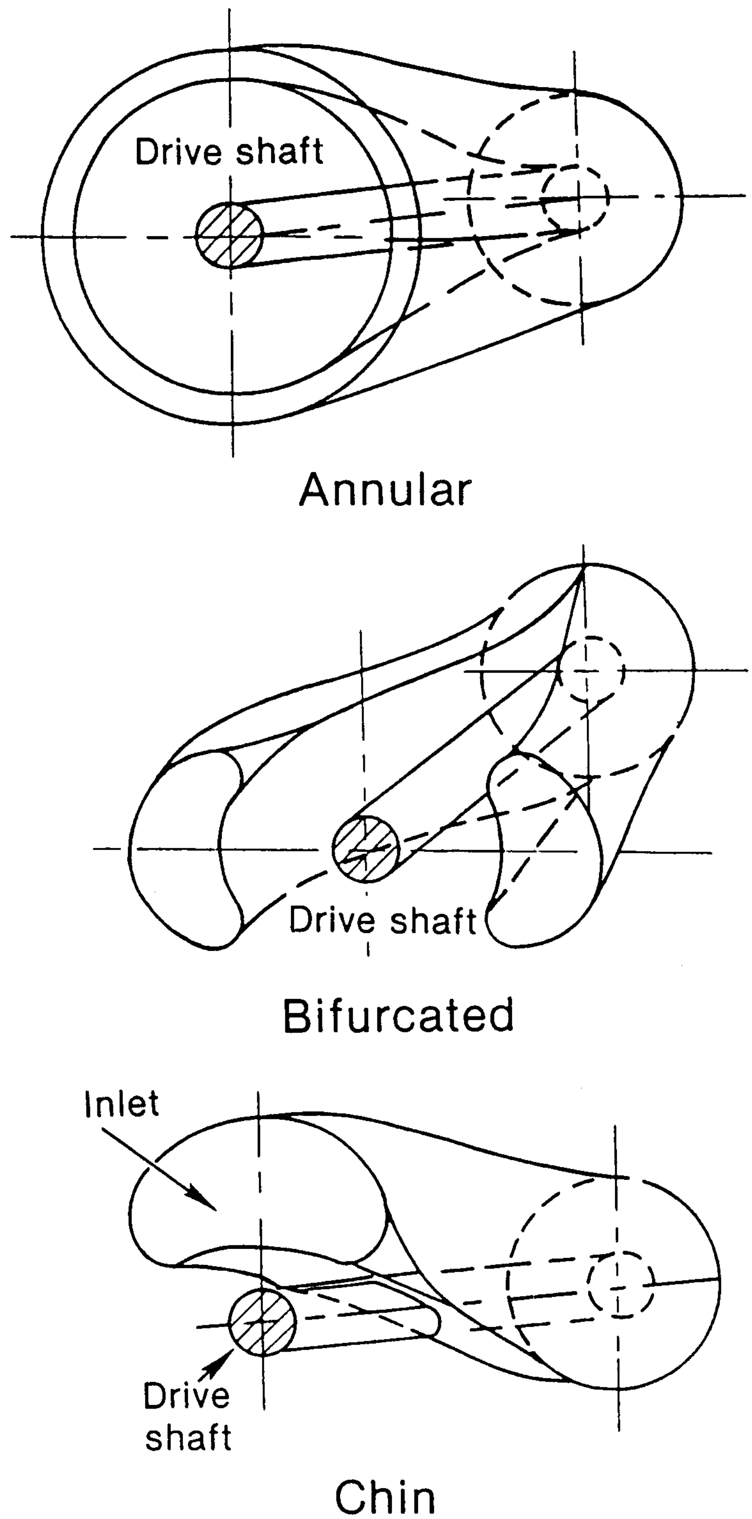

Figure 4.1-13 Candidate Inlets - The three candidate inlets are compatible with the reduction gear systems evaluated in the APET Program. (J27638-104)

4.1.6 Engine/Aircraft Trade Factors

Mission fuel burn trade factors were computed using a nominal domestic airline mission profile with step cruises and ATA reserves (see Figure 4.1-1). Trades were computed by changing one parameter (engine weight, for example) while holding all others at their baseline values and then running the airplane through the design mission analysis, including any resizing required to perform the specified design mission. Next, the resized airplane was run through the typical mission profile to compute its fuel burn. Results were then compared to the fuel burn of the baseline system, thus defining the sensitivity of fuel burn to, in this case, engine weight. Direct operating cost trade factors were computed in a similar fashion. Typical mission (740 km (400 nm)) trade factors are shown in Table 4.1-III. These trade factors were updated during Task IV (see Section 4.4).

TABLE 4.1-III
PROP-FAN AIRPLANE TRADE FACTORS
(1981 Dollars, $0.396 per liter ($1.50 per gallon))

Mach 0.75, 120 Passenger, Twin Engine Airplane
740 km (400 nm) Typical Mission
Sized for 10,668 m (35,000 ft) Cruise Altitude

Trade Factor	Effect on Fuel Burn	Effect on Direct Operating Cost
1% increase in TSFC	1.12%	0.46%
4 kg (10 lb) increase in pod drag per engine	0.32%	0.15%
450 kg (1000lb) increase in weight per engine	2.16%	1.29%
$100,000 increase in the price of each engine	---	0.27%
$10.00 increase in maintenance cost per engine flight hour	---	0.96%

4.1.7 Economic Considerations and Environmental Constraints

4.1.7.1 Economic Considerations

Fuel Price

A jet fuel price of $0.396 per liter ($1.50 per U.S. gallon), stated in 1981 dollars, was selected for the economic analysis. This is a representative mid-1990's level, assuming that fuel price escalates about 3% faster than general inflation. After final turboprop engine configurations had been defined, the economic impact of fuel prices of $0.264 and $0.528 per liter ($1.00 and $2.00 per gallon) were also assessed.

Direct Operating Cost Methods, Equations and Constraints

Direct Operating Cost (DOC) ground rules and equations are given in Table 4.1-IV. The method is based on the 1977 Boeing DOC method, updated to 1981 cost levels by Pratt & Whitney. The aircraft pricing equation was derived by Pratt & Whitney from published data. This method has been used in the Energy Efficient Engine program at 1977 and 1980 cost levels.

TABLE 4.1-IV
GROUND RULES AND EQUATIONS
(Direct Operating Cost Ground Rules)

Factor	Method of Calculation
Crew Cost	1981 update of 1977 Boeing
Fuel	\$0.396/liter (\$1.50/gallon) in 1981 dollars (The effect of \$0.264/liter (\$1.00/gallon) and \$0.528/liter (\$2.00/gallon) will also be evaluated.)
Aircraft	
o Price	Pratt & Whitney 1981
o Utilization	1981 update of 1977 Boeing
o Block Time	1981 update of 1977 Boeing
Insurance	1/2% flyaway per year
Airframe Maintenance	1981 update of 1977 Boeing
Maintenance Burden	200% on labor
Depreciation	Straight line, 15 years to 10% residual
Spares	Airframe and nacelle 6% Engine 30%
Engine Maintenance	Mature engine, no immaturity bump
Year dollars	1981
Crew Cost (2 man crew)*	Domestic = $(40.8F_W + 33.98)F_U + 39.1$ (F_W and F_U are from 1977 Boeing Method)
Airplane Price	
o Airframe	$1.1 \times 0.7079\ (\text{WAF}/1000)^{0.7} \times 10^6$
o Furnishings	$1.1 \times 1.4157\ (0.0089\ (\text{number of seats}) - 0.315) \times 10^6$
o Avionics	$1.1 \times 1.4157\ (0.0022\ (\text{number of seats}) + 1.81) \times 10^6$

TABLE 4.1-IV (Continued)

Economic Equations

Factor	Method of Calculation
Utilization	Constant trips/year as function of range (3200 at 463 km (250 nm), 2200 at 926 km (500 nm), 1400 at 1,852 km (1000 nm), 850 at 3,704 km (2000 nm)
Block time	Taxi Time - Domestic 14 minutes
Airframe Maintenance*	Material = 0.366 (WAF/1000)/Block Time + 0.294 (WAF/1000) Labor = $[0.07345\ (WAF/1000)^{0.7908}$/Block Time $+ 0.2048\ (WAF/100)^{0.595}]$ x Labor Rate where F_w is a function of airplane speed and gross weight F_u is a function of aircraft utilization WAF = Airframe Weight = OEW - Engine Weight Labor Rate (Direct) = $13.75/hour

*Cost in dollars per block hour

4.1.7.2 Environmental Constraints

Study engines were designed to meet or exceed existing noise and emissions regulations for new airplanes. Noise and emissions levels were documented to facilitate comparison with any projected regulations.

Noise

Noise was calculated according to FAR Part 36 procedures and compared to maximum acceptable levels. These maximum acceptable levels are functions of takeoff gross weight and number of engines, and were applied equally to the reference turbofan and the study turboprop engines.

Noise prediction procedures used in the Advanced Prop-Fan Engine Technology Study are described in detail in the Study Procedures and Assumptions Document (Reference 1). Airframe weight penalties for reducing cabin noise to acceptable levels through the use of advanced nacelle treatment techniques were based on work performed by the Lockheed-California Company (References 2 and 3).

Emissions

The emissions goals were those of the International Civil Aviation Organization (ICAO), referred to as "Research Goals," for newly certified engines. These goals are listed in Table 4.1-V.

TABLE 4.1-V
INTERNATIONAL CIVIL AVIATION ORGANIZATION
(Emissions Research Goals)

	Research Goals (g/kN)*	
	Turbofan	Prop-Fan
Unburned Hydrocarbons	4.35	4.35
Carbon Monoxide	42.0	42.0
Oxides of Nitrogen	56.6	54.0
Smoke (SAE Number)	24.7	24.4

* Thrust at Sea Level Takeoff Static Conditions in kilonewtons

These goals are basically for Class T2 turbofan engines (turbofan or turbojet engines with thrust greater than 35 kN (7870 lb)). In the past, turboprop engines have been treated differently and the units most often used were g/kN. For this APET study, the turboprop was treated as a turbofan when emissions were calculated. Thus, the same "Research Goals" were used for the turboprop and the reference turbofan and the units are the same (g/kN).

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SECTION 4.2 -- DISCUSSION OF RESULTS
Task II -- Cycle Optimization and Engine Configuration Selection

4.2 TASK II - CYCLE OPTIMIZATION AND ENGINE CONFIGURATION SELECTION

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List of Tables for Section 4.2 (Cont'd)

4.2 TASK II - CYCLE OPTIMIZATION AND ENGINE CONFIGURATION SELECTION

4.2.1 Introduction

The objectives of Task II were to identify the optimum cycle for an advanced turboprop engine and to select the most promising turboprop engine configurations for the integrated Prop-Fan propulsion system evaluation conducted under Task III of the APET Program.

The cycle optimization study considered a wide range of overall pressure ratios (from 20 to 50:1) and maximum combustor exit temperatures from 1204°C (2200°F) to 1537°C (2800°F). To determine the impact on size, the optimization study considered a base size of 16,000 shp, a small engine size of 8,000 shp and a large engine size of 23,000 shp. The key parameters used to select the cycle were fuel burn and direct operating cost. The optimum cycle for the base size 16,000 shp engine is 35:1 design overall pressure ratio and 39.5:1 at maximum climb, with a maximum combustor exit temperature of 1426°C (2600°F). The optimum cycle for the small size engine is 33:1 design overall pressure ratio and 37:1 at maximum climb, and 37:1 design overall pressure ratio and 41.5:1 at maximum climb for the large size engine, both of these at 1426°C (2600°F) maximum combustor exit temperature. The design point represents engine operation at typically 94% maximum cruise power.

Four candidate engine configurations were screened in the configuration selection process. They include: 1) a two-spool engine with all-axial compression, 2) a three-spool engine with axial/centrifugal compression, 3) a three-spool engine with all axial compression and 4) a three-spool engine like 2) with reversed flow where the inlet is at the rear and the turbine and propeller at the front. Fuel burn and direct operating cost were the parameters used to select the configuration. The two most promising configurations were 1) the two-spool with all axial compression and 3) the three-spool with axial/centrifugal compression.

The optimum cycle and the two configuration choices were approved by the NASA Program Manager and then used for the Task III Propulsion System Integration Study.

4.2.2 Cycle Optimization

The cycle optimization study represents the initial step in the engine definition process. The major objectives of the cycle optimization study were to:

- o Optimize the engine cycle for the reference aircraft, a 120-passenger airplane with a 3333 km (1800 nm) design range, a cruise altitude of 10,668 m (35,000 ft), and a cruise Mach number of 0.75. Both fuel burn and direct operating cost were figures of merit used in the optimization studies.

- o Investigate the impact of engine size on cycle selection by evaluating 8000 and 23,000 horsepower size engines in addition to the 16,000 horsepower size base engine.

To initiate the cycle studies, component performance and turbine cooling trends were established at levels appropriate for commercial engine certification in the 1992 time period.

Cycle studies were conducted for a base size turboprop engine (16,000 shaft horsepower) using the range of pressure ratios and combustor exit temperatures shown in Table 4.2-I. Each of the engine cycles in the matrix was evaluated on the basis of fuel burn and direct operating cost using trade factors defined in the study procedures and assumptions (Task I).

TABLE 4.2-I
CYCLE MATRIX FOR BASE SIZE ENGINE EVALUATION

Design Point Combustor Exit Temperature °C (°F)	Maximum Takeoff Combustor Exit Temperature °C (°F)	Design Point Pressure Ratio					
		20	25	30	35	40	45
971 (1780)	1204 (2200)	X	X	X	X	X	X
1071 (1960)	1315 (2400)	X	X	X	X	X	X
1121 (2050)	1371 (2500)	X	X	X	X	X	X
1176 (2150)	1426 (2600)	X	X	X	X	X	X
1271 (2320)	1537 (2800)	X	X	X	X	X	X

In addition to the cycle study for the base size engine, more limited studies were conducted for the 8000 and 23,000 shaft horsepower engines to determine the impact of engine size on selection of the optimum cycle. Fuel burn and direct operating cost characteristics were determined for these two engine sizes in the reference aircraft.

4.2.2.1 Cycle Optimization Study Ground Rules

A detailed set of ground rules was developed for the cycle optimization study (see Table 4.2-II). Use of these ground rules provided a consistent, objective evaluation of engine performance and minimized the impact of variables other than overall pressure ratio and maximum combustor exit temperature.

Maintaining constant propeller tip speed and power loading limited the impact of the Prop-Fan on the engine cycle. Previous work indicated that a tip speed of 243 m/sec (800 ft/sec) and power loading of 34 shp/D^2 was a reasonable design choice based on fuel burn and direct operating cost.

In order to insure a consistent philosophy of power-turbine work extraction for all cycles in the matrix, the design point primary stream jet velocity was held constant at 304 m/sec (1000 ft/sec). Once the optimum cycle was identified, a follow-on study of power turbine work extraction was conducted. This study confirmed that a velocity of 304 m/sec (1000 ft/sec) at the aerodynamic design point did provide the best combination of fuel burn and direct operating cost.

Holding thrust ratio constant at takeoff/climb and climb/cruise ensured that each engine would be rated consistently for the critical operating conditions. These thrust ratios were tailored to the requirements of the 120-passenger reference aircraft.

Selecting a consistent growth philosophy aided objective evaluation of engine performance. Ten percent growth in shaft horsepower reflects projected future applications for Prop-Fan propulsion.

The use of a constant aerodynamic overflow guaranteed that similar demands would be made on the turbomachinery of each engine. (Aerodynamic overflow is defined as the ratio of the inlet corrected flow at maximum climb power to the inlet corrected flow at the aerodynamic design point.) The variation in turbine cooling flows was required to achieve the same life for all cycles.

TABLE 4.2-II
GROUND Rules for Cycle Optimization

- o The study should reflect 1988 technology availability and 1992 engine certification.
- o Constant Propeller Tip Speed U_T = 243 m/sec (800 ft/sec) and Constant Propeller Loading (shp/D^2) = 34 at Maximum Climb Rating at a Mach Number of 0.75, 10,668 m (35,000 ft) Altitude
- o Constant Primary Stream Jet Velocity at Aerodynamic Design Point of 304 m/sec (1000 ft/sec)
- o Constant Thrust Ratios to Ensure Rating Consistency Between All Engines Studied:

  $$\frac{\text{Fn MCL, 10,668 m (35,000 ft), 0.75 M}}{\text{Fn T.O. M = 0.22 +14°C (+25°F)}} = 0.24$$
- o Constant Climb/Cruise Thrust Margin

  $$\frac{\text{Fn MCL, 10,668 m (35,000 ft), 0.75 M}}{\text{Fn MCR, 10,668 m (35,000 ft), 0.75 M}} = 1.09$$
- o Constant Growth Philosophy for All Cycles
  - 10% Takeoff Shaft Horsepower
- o Constant Overflow for All Cycles +10.5% W T2/ T2 Relative to Aerodynamic Design Point
- o Component Efficiencies Vary with Size
- o Turbine Cooling Varies with Maximum Combustor Exit Temperature (Takeoff) and Compressor Discharge Temperature (Takeoff). Turbine Cooling Set Based on Growth Ratings

4.2.2.2 Base Size Engine Evaluation

The base size engine (16,000 shp) was evaluated at each combination of overall pressure ratio and combustor exit temperature shown in Table 4.2-I. This large matrix covered cycles used in current engines as well as cycles projected for advanced technology engines of the 1990's. The detailed cycle study conducted for the base size engine investigated the effects of cycle parameters (pressure ratio and temperature) on turbine cooling trends, engine core size, thrust specific fuel consumption, engine, propeller, gearbox and overall propulsion system weight, fuel burn for a typical mission, and direct operating cost for a typical mission. Results of the evaluation are presented in this section.

Turbine Cooling Plus Leakage Flow Trends

Figure 4.2-1 shows how turbine cooling plus leakage flows varied with changes in overall pressure ratio and maximum combustor exit temperature. Increases in pressure ratio and temperature both led to increased cooling plus leakage flows. This increased flow was required to maintain constant turbine airfoil metal temperature, thereby ensuring engine durability.

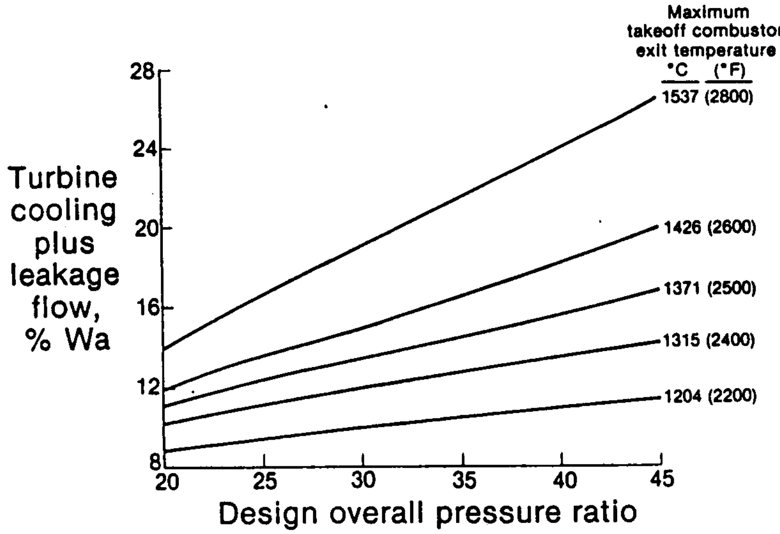

Figure 4.2-1 Effect of Cycle Parameters on Turbine Cooling Plus Leakage Flows - As pressure ratio or temperature increases, cooling flow must increase in order to maintain constant turbine airfoil metal temperature for durability. (J27638-38)

Core Size

The effect of cycle parameters on engine core size (high compressor exit corrected airflow) is shown in Figure 4.2-2. As overall pressure ratio increases at constant maximum combustor exit temperature, the core size decreases. At constant overall pressure ratio, the aerodynamic core size decreases as maximum combustor exit temperature increases. The smaller size core results in a reduction in efficiency for the smaller size engine components.

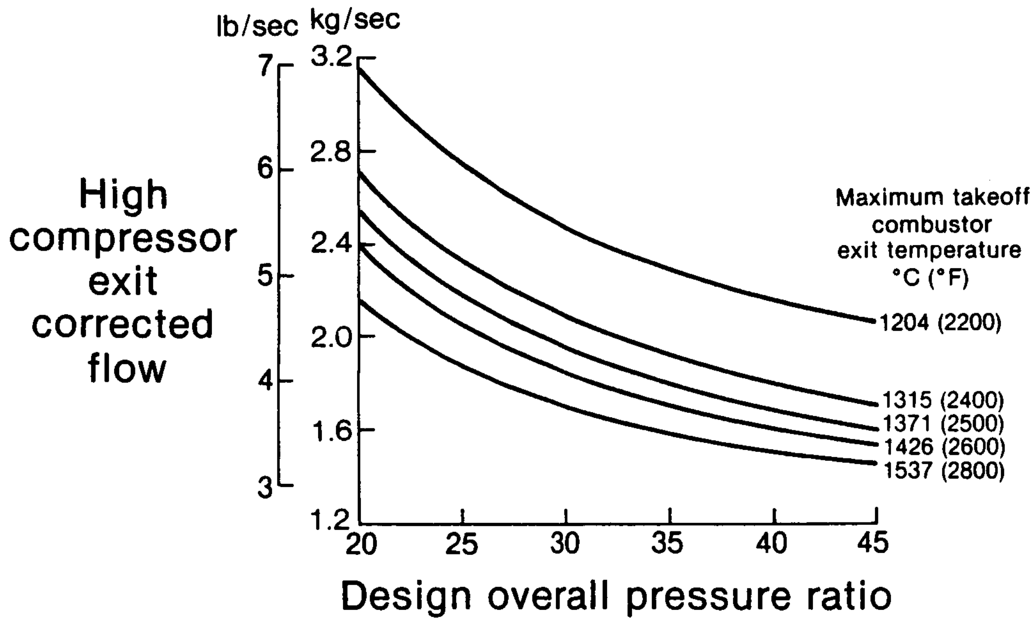

Figure 4.2-2 Effect of Cycle Parameters on Core Size - As overall pressure ratio or maximum combustor exit temperature increases, engine core size decreases. (J27638-39)

Engine TSFC

Figure 4.2-3 shows the resulting variation in thrust specific fuel consumption with changes in overall pressure ratio and maximum combustor exit temperature. The best installed thrust specific fuel consumption is achieved at maximum combustor exit temperatures between 1371°C (2500°F) and 1426°C (2600°F) regardless of overall pressure ratio. In this temperature range, thrust specific fuel consumption reaches a minimum at overall pressure ratios of 40 to 45 or greater. Maximum combustor exit temperatures above 1426°C (2600°F) do not improve thrust specific fuel consumption at any overall pressure ratio of interest because of the increasing amount of cooling plus leakage flow which must accompany the use of higher temperatures.

Engine Weight

Figure 4.2-4 shows engine weight as a function of overall pressure ratio and maximum combustor exit temperature. Engine weight increases with rising overall pressure ratio and constant maximum combustor exit temperature. This increase in weight is due to the greater number of stages required to achieve the higher pressure ratio and an increasing air flow with overall pressure ratio. However, engine weight decreases with increasing maximum combustor exit temperature and constant overall pressure ratio. This decrease is due to the reduced airflow requirement associated with increased combustor exit temperatures.

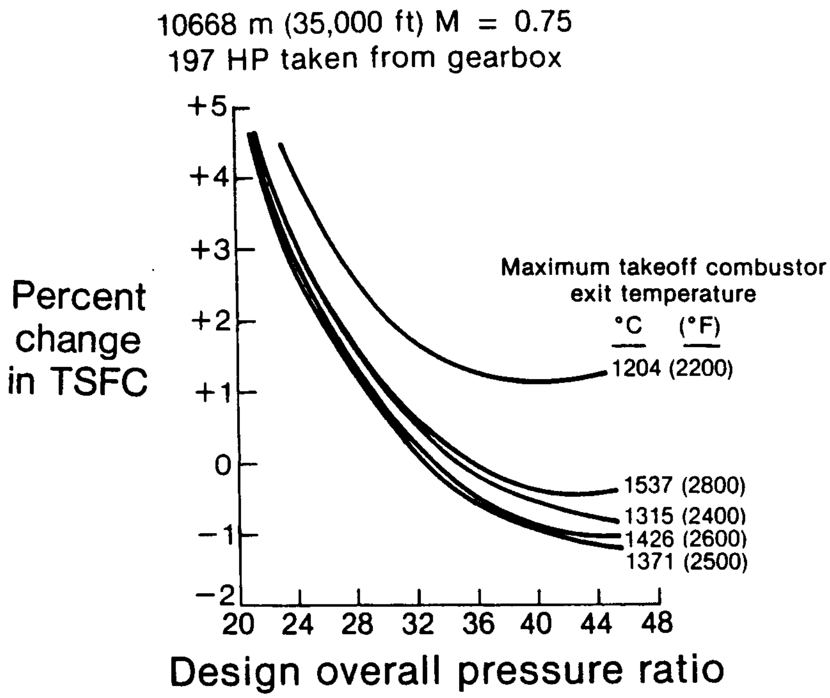

Figure 4.2-3 Effect of Cycle Parameters on Engine Cruise TSFC - The best cruise TSFC is achieved at 44:1 overall pressure ratio and 1371°C (2500°F) maximum combustor exit temperature. (J27638-40)

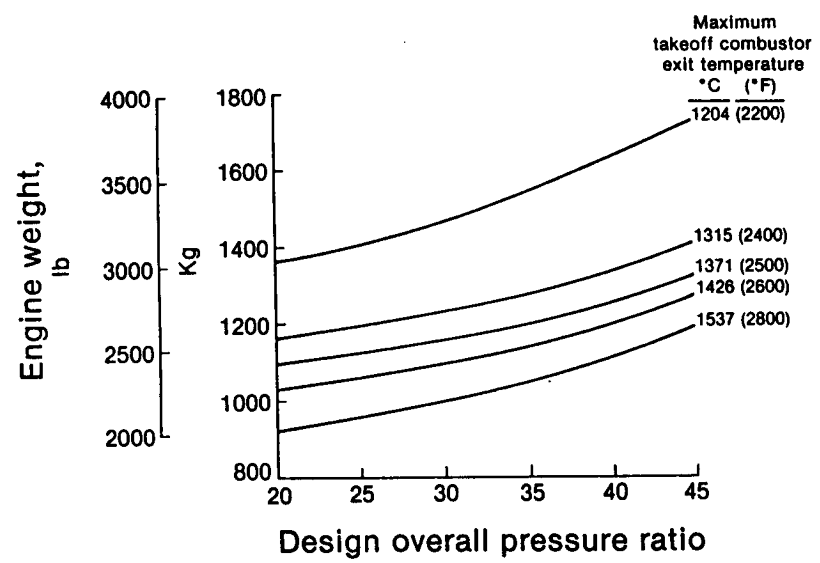

Figure 4.2-4 Effect of Cycle Parameters on Engine Weight - The requirement for more stages causes engine weight to increase as overall pressure ratio increases. However, as temperatures increase, engine size is reduced and engine weight decreases. (J27638-41)

Propeller Weight

The effect of cycle parameters on propeller weight is shown in Figure 4.2-5. Propeller weight decreases slightly as overall pressure ratio increases and combustor exit temperature decreases.

The reduction in propeller weight at higher pressure ratios results from a trend toward increased engine airflow as overall pressure ratio increases. As engine airflow increases, a greater amount of thrust is provided by the primary stream. Since total thrust is held constant, propeller thrust can be reduced slightly by using a smaller diameter unit, thus reducing propeller weight.

The reduction in propeller weight at lower combustor exit temperatures also results from increased engine airflow. As maximum combustor exit temperature is reduced, the engine airflow must increase, providing more primary stream thrust. Again propeller diameter can be reduced, thereby decreasing propeller weight at lower combustor exit temperatures.

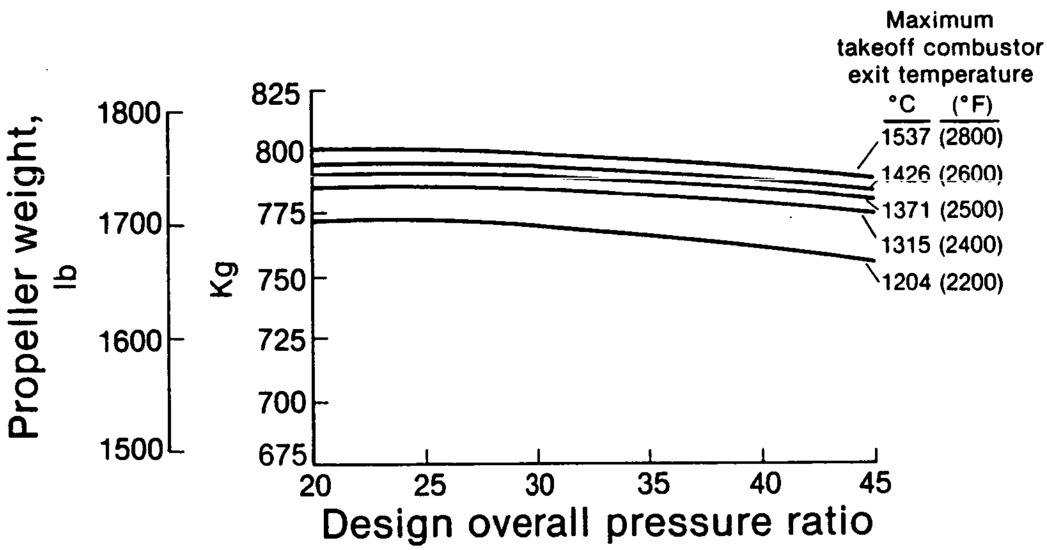

Figure 4.2-5 Effect of Cycle Parameters on Propeller Weight - Propeller weight remains nearly constant over the range of pressure ratios evaluated and varies only slightly as combustor exit temperature increases. (J27638-42)

Gearbox Weight

The effect of cycle parameters on gearbox weight is shown in Figure 4.2-6. Gearbox weight decreases slightly as overall pressure ratio increases and maximum combustor exit temperature decreases. This trend reflects the reductions in propeller diameter and required shaft horsepower which result from increased engine airflow under these conditions.

Gearbox weights are based on an offset gearbox using a compound idler arrangement.

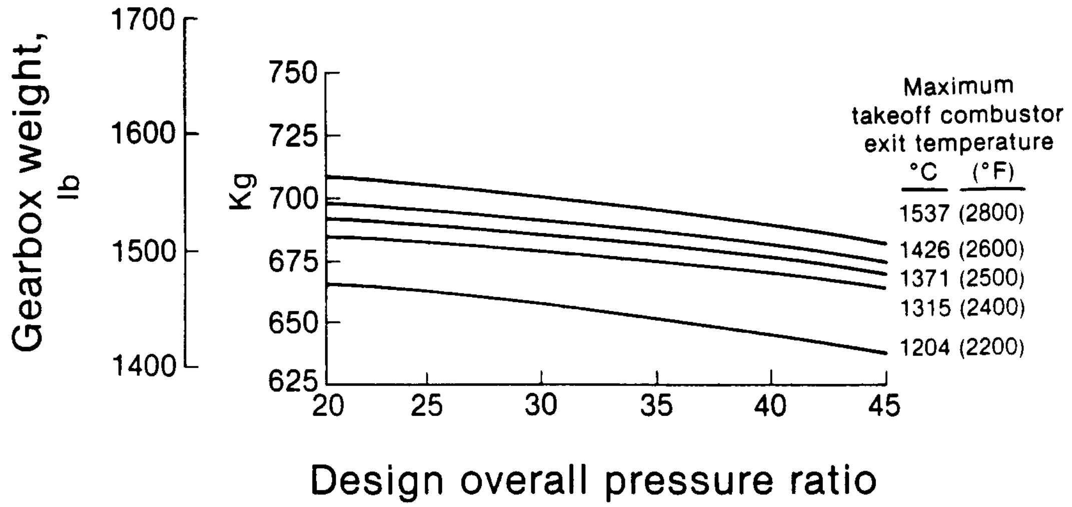

Figure 4.2-6 Effect of Cycle Parameters on Gearbox Weight - Gearbox weight remains nearly constant over the range of pressure ratios evaluated and varies only slightly as combustor exit temperature increases. (J27638-43)

Propulsion System Weight

Figure 4.2-7 shows how total propulsion system weight is affected by changes in the engine cycle. The total weight of the propulsion system increases as overall pressure ratio increases and maximum combustor exit temperature decreases. The total propulsion system includes the engine, propeller, gearbox, and nacelle.

Changes in engine weight are the major factor in the propulsion system weight trends. A change of 210 kg (463 lb) per engine is required to produce a 1% change in fuel burn over a typical mission.

The changes in total propulsion system weight have been arbitrarily normalized for a 1371°C (2500°F) maximum combustor exit temperature 30:1 overall pressure ratio engine. Nacelle weight is not shown because it is essentially constant.

Fuel Burned for a Typical Mission

Aircraft fuel burn trends for a typical 740 km (400 nm) mission are presented in Figure 4.2-8. The best fuel burn results are achieved at a maximum combustor exit temperature of 1426°C (2600°F). At this temperature, the best fuel burn is obtained at an overall pressure ratio of about 40:1.

Aircraft fuel burn optimizes at a lower pressure ratio and higher temperature than specific fuel consumption because of the effect of total propulsion system weight on aircraft weight.

Fuel burn results have been arbitrarily normalized to the 1371°C (2500°F) maximum combustor exit temperature 30:1 overall pressure ratio engine.

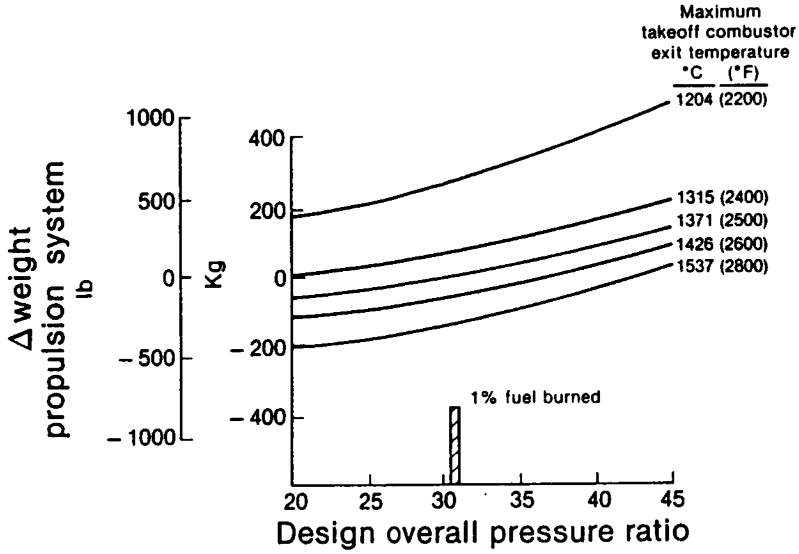

Figure 4.2-7 Effect of Cycle Parameters on Total Propulsion System Weight - Engine weight trends have the most significant impact on the propulsion system. A weight change of 210 kg (463 lb) per engine is required to produce a one percent change in fuel burn on an average mission. (J27638-44)

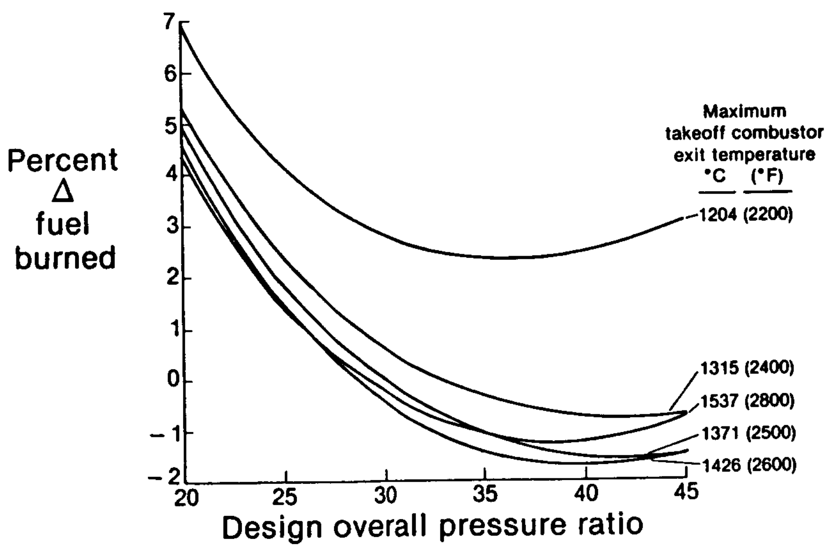

Figure 4.2-8 Effect of Cycle Parameters on Fuel Burned for a Typical Mission - The best fuel burn is achieved at a maximum combustor exit temperature of 1426°C (2600°F) and an overall pressure ratio of approximately 40:1. (J27638-45)

Direct Operating Costs for a Typical Mission

The direct operating cost results for a typical 740 km (400 nm) mission are shown in Figure 4.2-9. Direct operating costs are lowest at an overall pressure ratio of about 35:1 and a maximum combustor exit temperature of 1537°C (2800°F). However, direct operating costs are only 0.08% higher at an overall pressure ratio of 35:1 and a maximum combustor exit temperature of 1426°C (2600°F). This is judged to be a small penalty for 111°C (200°F) lower temperature operation and is well within the accuracy band.

Optimum Cycle for the 16,000 Horsepower Size Engine

Based on fuel burn and direct operating cost trends, the optimum cycle for a base size turboprop engine (16,000 shaft horsepower) is 35:1 at the design point and 1426°C (2600°F) maximum combustor exit temperature. A more detailed description of the cycle is presented in Table 4.2-III.

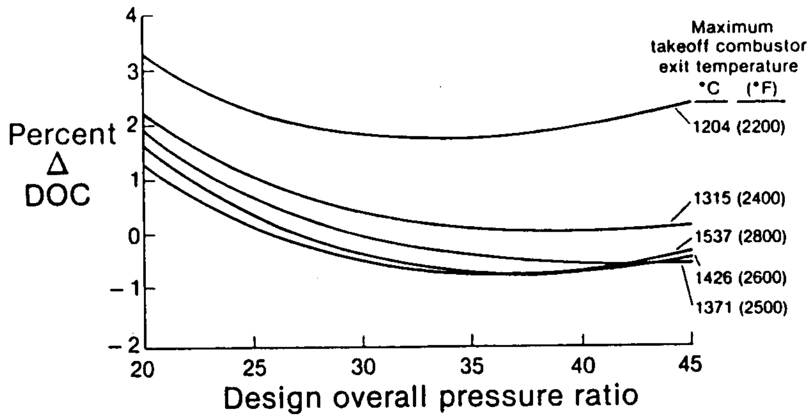

Figure 4.2-9 Effect of Cycle Parameters on Direct Operating Cost - Direct operating cost is lowest at an overall pressure ratio of about 35:1. A maximum combustor exit temperature of 1426°C (2600°F) provides acceptable operating costs without compromising engine performance. (J27638-46)

TABLE 4.2-III
OPTIMUM CYCLE FOR THE BASE SIZE ENGINE

Design Point Overall Pressure Ratio (at 90% Maximum Cruise Thrust)	35
Maximum Climb Overall Pressure Ratio	39.5
Maximum Cruise Overall Pressure Ratio	37.5
High Compressor Exit Corrected Flow	1.70 kg/sec (3.75 lb/sec)
Takeoff Combustor Exit Temperature	
Initial	1387°C (2530°F)
Growth	1426°C (2600°F)

4.2.2.3 Small Size Engine Evaluation

The 8000 horsepower size engine was evaluated at overall pressure ratios from 25:1 to 45:1 and maximum combustor exit temperatures of 1371°C (2500°F), 1426°C (2600°F), and 1482°C (2700°F). Based on study results for the base size engine, higher and lower combustor exit temperatures were not considered. The range of overall pressure ratios covers current engine operating conditions as well as projected operating conditions for engines of the 1990's. Specific fuel consumption, propulsion system weight, fuel burn, and direct operating cost trends were evaluated.

The cycle for the 8000 horsepower size engine was optimized at a design overall pressure ratio of 33:1, 37:1 at maximum climb, and a maximum combustor exit temperature of 1426°C (2600°F).

Engine TSFC

Figure 4.2-10 shows the variation in installed thrust specific fuel consumption with changes in overall pressure ratio and maximum combustor exit temperature. The best installed thrust specific fuel consumption is achieved at a maximum combustor exit temperature of 1371°C (2500°F) and an overall pressure ratio of 42:1. Maximum combustor exit temperatures above 1426°C (2600°F) do not improve thrust specific fuel consumption at any overall pressure ratio of interest because of the increasing amount of turbine cooling flow required for operation at higher temperatures.

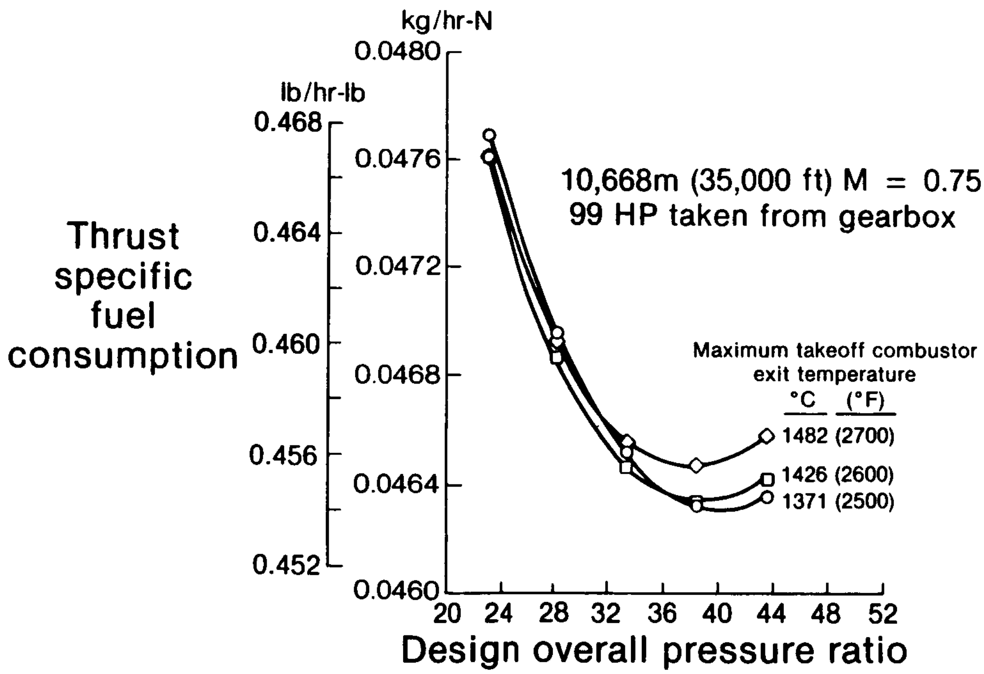

Figure 4.2-10 Effect of Cycle Parameters on Small Engine TSFC - Maximum combustor exit temperatures of 1371°C (2500°F) and overall pressure ratio of 42 provide minimum thrust specific fuel consumption. (J27638-47)

Propulsion System Weight

Figure 4.2-11 shows how total propulsion system weight is affected by changes in overall pressure ratio and maximum combustor exit temperature. The total propulsion system includes the engine, propeller, gearbox, and nacelle.

The variations in propulsion system weight for the 8000 horsepower size engine are generally similar to the trends observed with the base size engine (Section 4.2.1.2.7). However, the magnitudes of the variations are reduced because this engine is only half the power of the base engine.

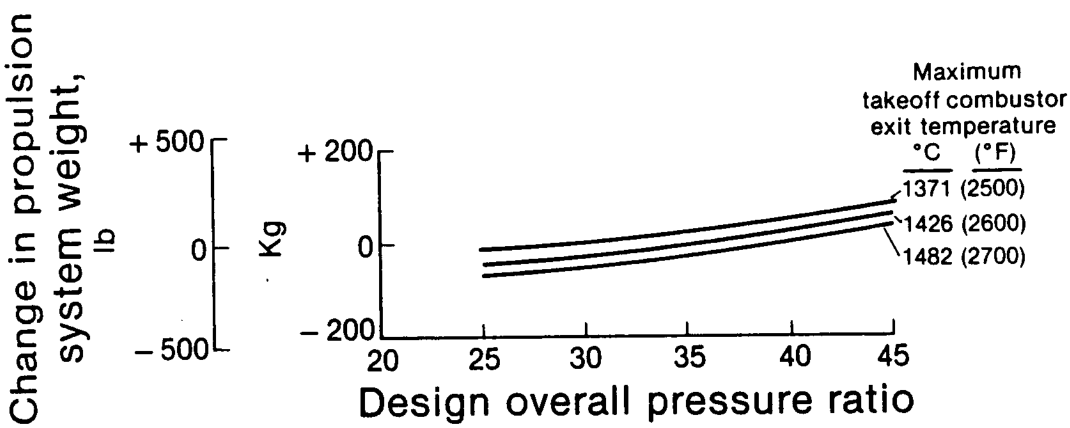

Figure 4.2-11 Effect of Cycle Parameters on Propulsion System Weight - Propulsion system weight trends for the 8000 horsepower engine parallel those for the base engine, but the variations are less pronounced due to smaller engine size. (J27638-48)

Fuel Burned for a Typical Mission

Fuel burn trends for a typical 740 km (400 nm) mission are presented in Figure 4.2-12. The best fuel burn results are achieved at a maximum combustor exit temperature of 1426°C (2600°F) and an overall pressure ratio of about 37:1.

Direct Operating Cost for a Typical Mission

The direct operating cost results for a typical 740 km (400 nm) mission are shown in Figure 4.2-13. The best direct operating costs are attained at a maximum combustor exit temperature of 1482°C (2700°F). However, direct operating costs are only slightly higher at a combustor exit temperature of 1426°C (2600°F). This is judged to be a small penalty for 56°C (100°F) lower temperature operation.

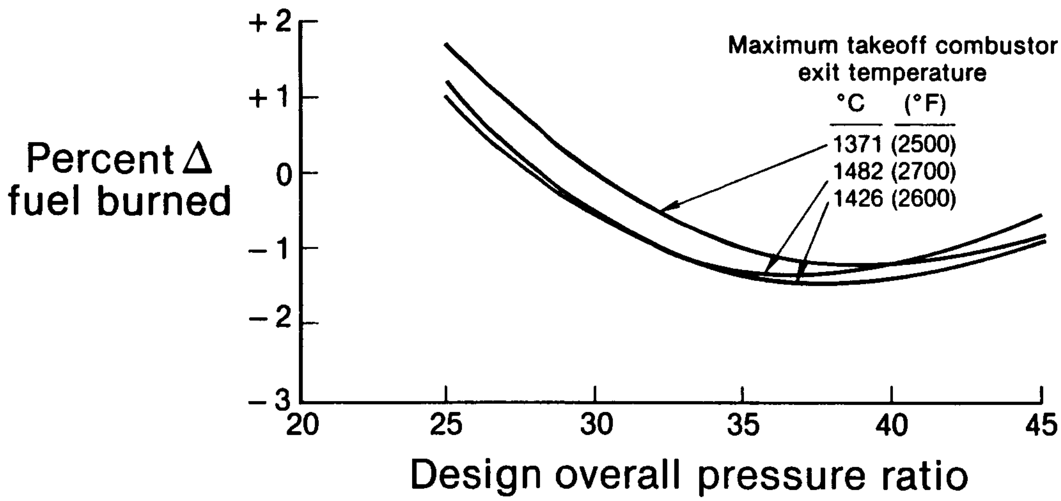

Figure 4.2-12 Effect of Cycle Parameters on Fuel Burned for a Typical Mission - The best fuel burn is achieved at a maximum combustor exit temperature of 1426°C (2600°F) and an overall pressure ratio of approximately 37:1. (J27638-49)

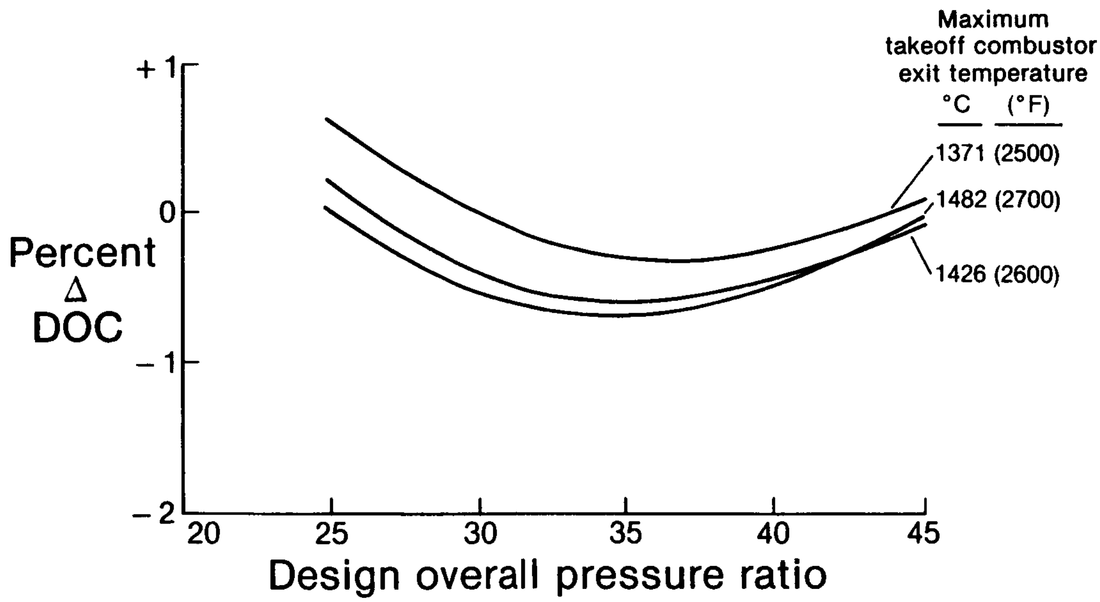

Figure 4.2-13 Effect of Cycle Parameters on Direct Operating Cost for a Typical Mission - Direct operating cost is lowest at a maximum combustor exit temperature of about 1482°C (2700°F) and an overall pressure ratio of about 33:1. (J27638-50)

Optimum Cycle for the 8000 Horsepower Size Engine

Based on fuel burn and direct operating cost trends, the optimum cycle for a small size turboprop engine (8000 shaft horsepower) is 33:1 design overall pressure ratio and 1426°C (2600°F) maximum combustor exit temperature. A more detailed description of the cycle is presented in Table 4.2-IV.

TABLE 4.2-IV
OPTIMUM CYCLE FOR THE SMALL SIZE ENGINE

Design Point Overall Pressure Ratio (at 90% Maximum Cruise Thrust)	33
Maximum Climb Overall Pressure Ratio	37.2
Maximum Cruise Overall Pressure Ratio	35.2
High Compressor Exit Corrected Flow	0.9 kg/sec (2.0 lb/sec)
Takeoff Combustor Exit Temperature	
Initial Rating	1387°C (2530°F)
Growth Rating	1426°C (2600°F)

4.2.2.4 Large Size Engine Evaluation

The 23,000 horsepower size engine was evaluated at overall pressure ratios from 30:1 to 50:1 and maximum combustor exit temperatures of 1371°C (2500°F), 1426°C (2600°F), and 1482°C (2700°F). Based on study results for the base size engine, higher and lower combustor exit temperatures were not considered. The range of overall pressure ratios covers current engine operating conditions as well as projected operating conditions for engines of the 1990's. Specific fuel consumption, propulsion system weight, fuel burn, and direct operating cost trends were evaluated.

The cycle for the 23,000 horsepower size engine was optimized at a design overall pressure ratio of 37:1 and a maximum combustor exit temperature of 1426°C (2600°F).

Engine TSFC

Figure 4.2-14 shows the variation in thrust specific fuel consumption with changes in overall pressure ratio and maximum combustor exit temperature. The best installed thrust specific fuel consumption is achieved at a maximum combustor exit temperature of 1371°C (2500°F) and an overall pressure ratio of about 45:1.

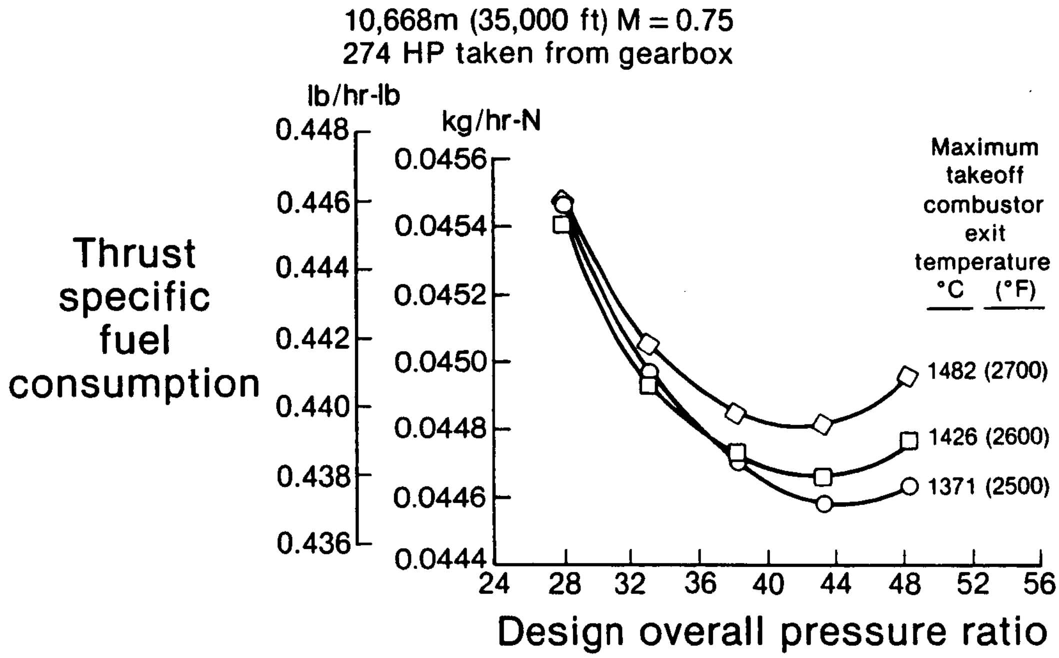

Figure 4.2-14 Effect of Cycle Parameters on Large Engine TSFC - The best thrust specific fuel consumption is obtained at combustor exit temperatures of 1371°C (2500°F) and an overall pressure ratio of about 45:1. (J27638-51)

Propulsion System Weight

Figure 4.2-15 shows how total propulsion system weight is affected by changes in overall pressure ratio and maximum combustor exit temperature. The total propulsion system includes the engine, propeller, gearbox, and nacelle.

The variations in propulsion system weight for the 23,000 horsepower size engine are generally similar to the trends observed with the base size engine (Section 4.2.1.2.7). The magnitude of the variations is increased because the engine is significantly larger.

Fuel Burned for a Typical Mission

Fuel burn trends for a typical 740 km (400 nm) mission are presented in Figure 4.2-16. The best fuel burn results are achieved at a maximum combustor exit temperature of 1426°C (2600°F) and an overall pressure ratio of about 42:1.

Direct Operating Costs for a Typical Mission

The direct operating cost results for a typical 740 km (400 nm) mission are shown in Figure 4.2-17. The best direct operating costs are attained at a maximum combustor exit temperature of 1426°C (2600°F) and a design overall pressure ratio of about 37:1.

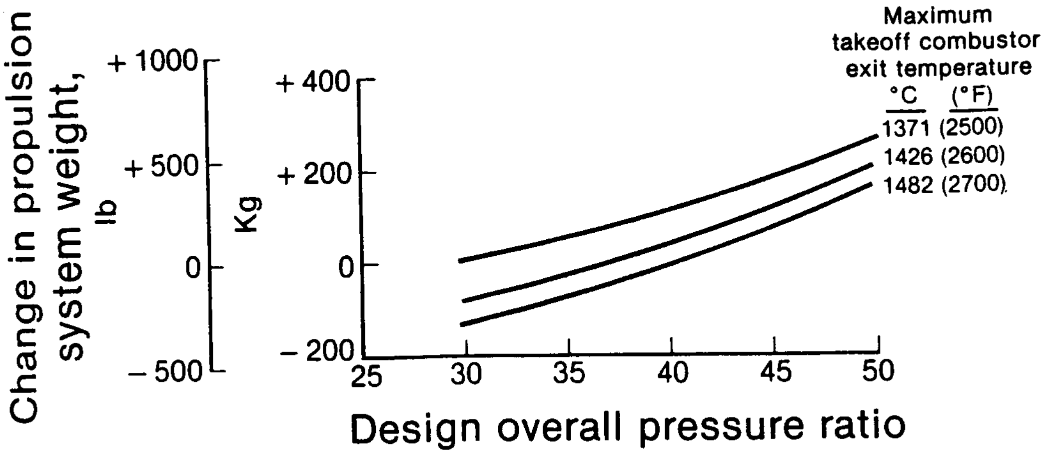

Figure 4.2-15 Effect of Cycle Parameters on Propulsion System Weight - Propulsion system weight trends for the 23,000 horsepower engine parallel those for the base engine, but the weight increments are larger due to increased engine size. (J27638-52)

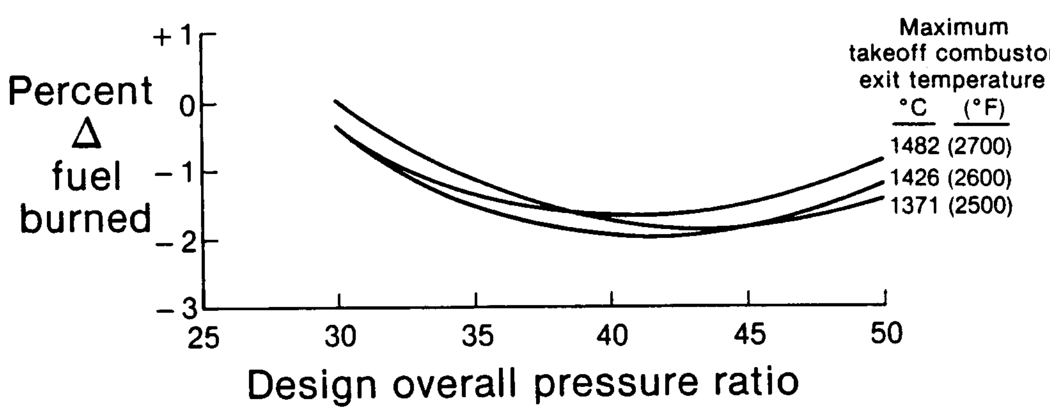

Figure 4.2-16 Effect of Cycle Parameters on Fuel Burned for a Typical Mission - The best fuel burn is achieved at a maximum combustor exit temperature of 1426°C (2600°F) and an overall pressure ratio of about 42:1. (J27638-53)

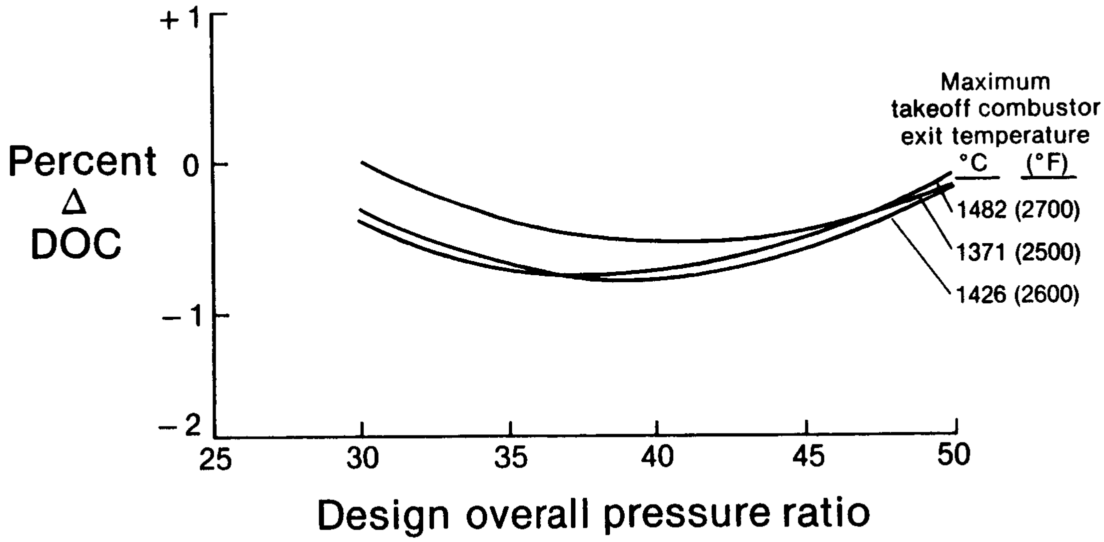

Figure 4.2-17 Effect of Cycle Parameters on Direct Operating Cost for a Typical Mission - Direct operating cost is lowest at a maximum combustor exit temperature of 1426°C (2600°F) and an overall pressure ratio of about 37:1. (J27638-54)

Optimum Cycle for the 23,000 Horsepower Size Engine

Based on fuel burn and direct operating cost trends, the optimum cycle for a large size turboprop engine (23,000 shaft horsepower) is 37:1 design overall pressure ratio and 1426°C (2600°F) maximum combustor exit temperature. A more detailed description of the cycle is presented in Table 4.2-V.

TABLE 4.2-V
OPTIMUM CYCLE FOR THE LARGE SIZE ENGINE

Design Point Overall Pressure Ratio (at 90% Maximum Cruise Thrust)	37
Maximum Climb Overall Pressure Ratio	41.5
Maximum Cruise Overall Pressure Ratio	39.5
High Compressor Exit Corrected Flow	2.24 kg/sec (4.96 lb/sec)
Takeoff Combustor Exit Temperature	
Initial Rating	1387°C (2530°F)
Growth Rating	1426°C (2600°F)

4.2.2.5 Summary of Cycle Optimization Study Results

Evaluation of 8000, 16,000 and 23,000 horsepower engines indicated that engine size had little impact on the selection of the optimum cycle (see Figure 4.2-18). In the 8000 to 23,000 horsepower range, the best overall pressure ratio is 33:1 and 37:1 at the design point. Balancing fuel burn and direct operating costs, the best combustor exit temperature is 1426°C (2600°F) regardless of engine size.

The overall pressure ratio is also shown at the maximum climb condition of 10,668 m (35,000 ft) Mach 0.75.

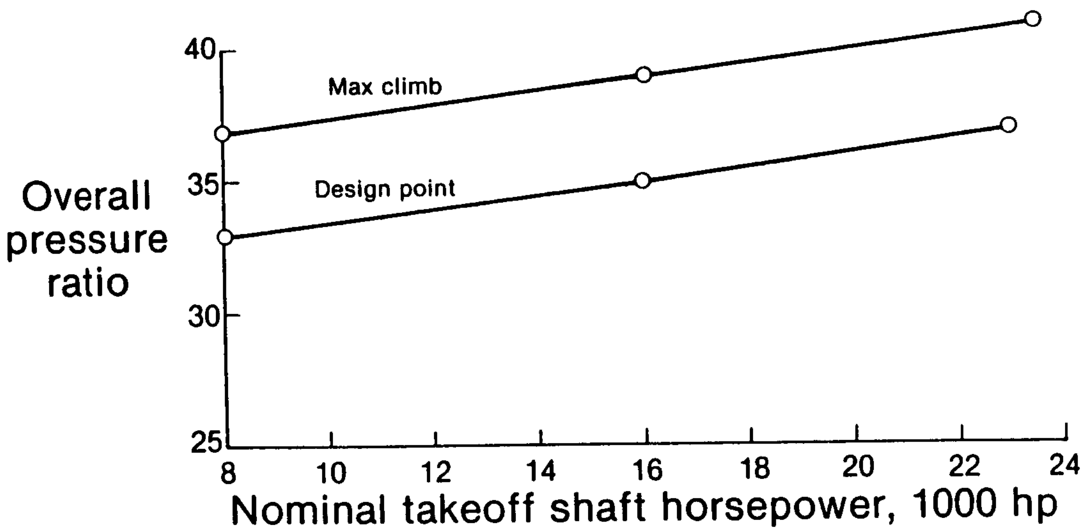

Figure 4.2-18 Effect of Size on Cycle Pressure Ratio - For the range of engine sizes evaluated, the optimum overall pressure ratio varies between 33:1 and 37:1 at design point. The optimum maximum combustor exit temperature is 1426°C (2600°F) regardless of engine size. (J27638-60)

A more detailed description of the optimum cycle for the base 16,000 shp engine is presented in Table 4.2-VI. This cycle was subsequently used in the detailed evaluation of the two most promising turboprop engine configurations.

TABLE 4.2-VI
CHARACTERISTICS OF THE OPTIMUM ENGINE CYCLE

	Initial Rating	Growth Rating
Aerodynamic Design Point		
Shaft Horsepower	6470	6470
Overall Pressure Ratio	35.0	35.0
Combustor Exit Temperature, °C (°F)	1177 (2150)	1177 (2150)
Sea Level Takeoff M = 0.3 Std +14°C (+25°F)		
Shaft Horsepower	16,550	18,200
Overall Pressure Ratio	30.9	32.9
Combustor Exit Temperature, °C (°F)	1387 (2530)	1426 (2600)
Maximum Climb at 10,668 m (35,000 ft) M = 0.75		
Shaft Horsepower	7400	7840
Overall Pressure Ratio	39.5	41.2
Combustor Exit Temperature, °C (°F)	1254 (2290)	1287 (2350)
Maximum Cruise at 10,688 m (35,000 ft) M = 0.75		
Shaft Horsepower	6880	7290
Overall Pressure Ratio	37.5	39.0
Combustor Exit Temperature, °C (°F)	1221 (2230)	1246 (2275)

4.2.3 Engine Configuration Evaluation

To explore a variety of approaches to engine design, four candidate engine configurations were screened in a 10,000 horsepower base engine size and a reference cycle. This screening covered mechanical design, performance, design assurance and environmental issues. The two-spool all-axial compression engine and the three-spool axial/centrifugal compression engines emerged as the best configurations for a Prop-Fan powered aircraft. The three-spool all-axial configuration was inferior in deteriorated performance and was considered a high technical risk. The reversed engine configuration was inferior in fuel burn and high in installation complexity.

After the screening was completed, the best engine configurations were updated using the cycle characteristics identified in the optimization study (Section 4.2.2). The engines were reevaluated on the basis of mechanical design, performance, design assurance, and environmental issues. Although there were differences in fuel burn and direct operating costs, the two engines were considered approximately equal. Thus, both the two-spool axial and three-spool axial/centrifugal engines were carried into the propulsion system integration studies conducted with the airframe manufacturers (Task III of the APET Program).

4.2.3.1 Configuration Study Objectives and Ground Rules

Study Objectives

The major objectives of the configuration study were to:

- o Determine whether a centrifugal compressor has a place in the 10,000 to 15,000 horsepower size turboprop engines being evaluated for the reference aircraft.
- o Evaluate the differences between two-spool and three-spool turboprop engines.
- o Determine whether a non-concentric shaft turboprop engine results in a system benefit. (A non-concentric shaft turboprop engine is defined as an engine concept in which the power turbine shaft is not required to pass through the middle of the turboprop engine.) This concept was explored in the reversed three-spool axial/centrifugal compression engine.

Study Ground Rules

Four candidate engine configurations were considered. A detailed set of study parameters was developed in order to ensure a comprehensive, objective evaluation of engine design and performance characteristics. A standard engine size and cycle were selected to screen the candidates. The most promising configurations were then updated using the optimum turboprop engine cycle (discussed in Section 4.2.1).

Turboprop Engine Configuration Candidates - The four turboprop engine configurations evaluated in the APET Program are shown in Figure 4.1-7. They include a two-spool all-axial engine configuration (designated STS648), a three-spool all-axial engine configuration (designated STS647), a three-spool axial/centrifugal engine configuration (designated STS646), and the reversed axial/centrifugal engine configuration (designated STS646R).

In the two-spool axial configuration (Figure 4.1-7A), the power turbine drives both the low-pressure compressor and the Prop-Fan. The two three-spool configurations (Figures 4.1-7B and 4.1-7C) permitted evaluation of free power turbines relative to the two-spool non-free turbine configuration. These two configurations were also used to evaluate the relative merits of axial versus axial/centrifugal compressors. The "reversed" engine configuration (Figure 4.1-7D) provided the capability to explore an installation arrangement which has a free turbine without a third concentric shaft.

Engine configuration candidates with the entire compression system on one spool were evaluated in previous Pratt & Whitney studies and found not to be competitive. Therefore, neither an all axial or axial/centrifugal one spool configuration was evaluated in the APET Program.

Configuration Evaluation Parameters - A comprehensive set of evaluation parameters was developed for the configuration selection process (see Table 4.2-VII). These parameters can be divided into four major categories: (1) mechanical design and analysis related issues; (2) performance related issues; (3) design assurance related issues; (4) environmental issues (noise and pollution).

TABLE 4.2-VII
ENGINE CONFIGURATION EVALUATION PARAMETERS

Mechanical Design and Analysis Related Issues

Turbine Cooling Requirements
Component Performance and Matching
Future Engine Growth Paths
Deterioration Modes and Design Considerations
Maintenance Considerations (Modularity)
Materials Selection Based on Structural Design Limits
Timeliness of Technology
Impact of Engine Design Choices on Propeller, Gearbox, Accessories, and Oil Cooling

Performance Related Issues

Specific Fuel Consumption
Fuel Burned on Typical Mission
Direct Operating Costs on Typical Mission
Overall Pressure Ratio
Turbine Rotor Inlet Temperature
Power Turbine Work Extraction
Starting Requirements
Off-Design Operations
Propeller Drag in Connection with Failure Modes
Effects of Anticipated Customer Bleeds and/or Horsepower Extraction
Component Performance and Matching
Future Engine Growth Paths
Operating Constraints
Impact of Engine Design Choices on Propeller, Gearbox, Accessories, and Oil Cooling

Design Assurance Related Issues

Engine Weight
Relative Engine Cost
Maintenance Considerations (Relative Maintenance Cost)
Engine Reliability

Environmental Related Issues (Noise and Pollution)

Engine Noise Considerations
Engine Emissions Considerations

Engine Size - In Figure 4.2-19, engine size (shaft horsepower) is plotted against aircraft size (number of passengers). The top portion of the band represents a cruise Mach number of 0.8 and a cruise altitude of 10,668 m (35,000 ft); the bottom portion of the band represents a cruise Mach number of 0.7 and a cruise altitude of 9448 m (31,000 ft). All studies conducted to date by both engine and airframe manufacturers indicate that the power requirements for a Prop-Fan powered aircraft fall within this band.

Studies conducted by Pratt & Whitney indicate that as engine size is reduced, bearing compartments, rotor dynamics, and combustor aerothermal mechanical conditions become critical factors in the mechanical design of the engine. Thus the 10,000 horsepower engine size (at sea level static conditions) was used to screen the four engine configurations.

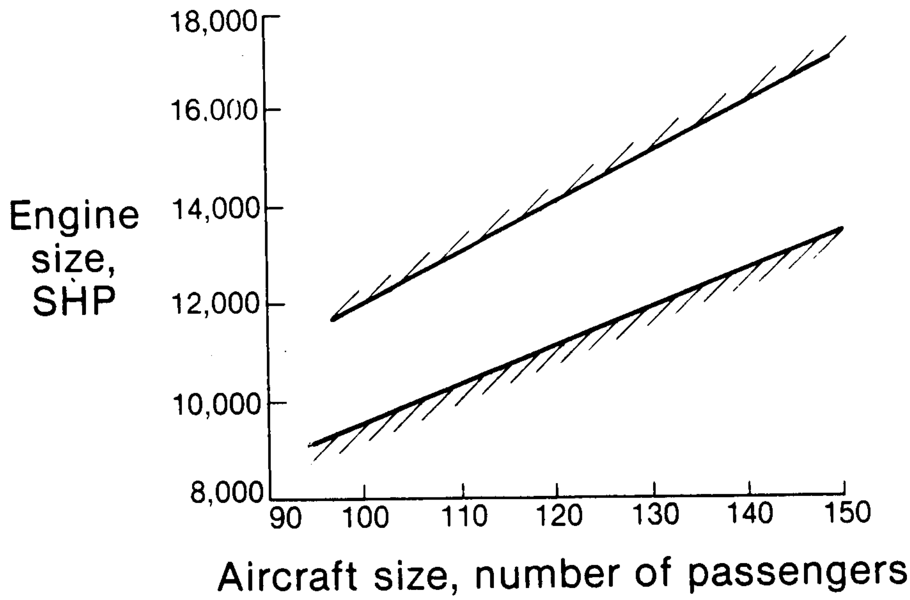

Figure 4.2-19 Shaft Horsepower vs Passengers - The engine size for a 100 - 150 passenger short to medium range transport is likely to be between 9000 and 16,000 shaft horsepower. (J27638-55)

In Figure 4.2-20 lines of constant high pressure compressor exit corrected airflow are superimposed on Figure 4.2-19. At flow rates above 1.8 kg/sec (4.0 lb/sec) axial compression systems have been used exclusively. Below 0.9 kg/sec (2.0 lb/sec), experience encompasses both axial and centrifugal compression systems. The configuration screening was conducted at a flow rate of slightly more than 0.9 kg/sec (2.0 lb/sec) to assess both axial and axial/centrifugal configurations.

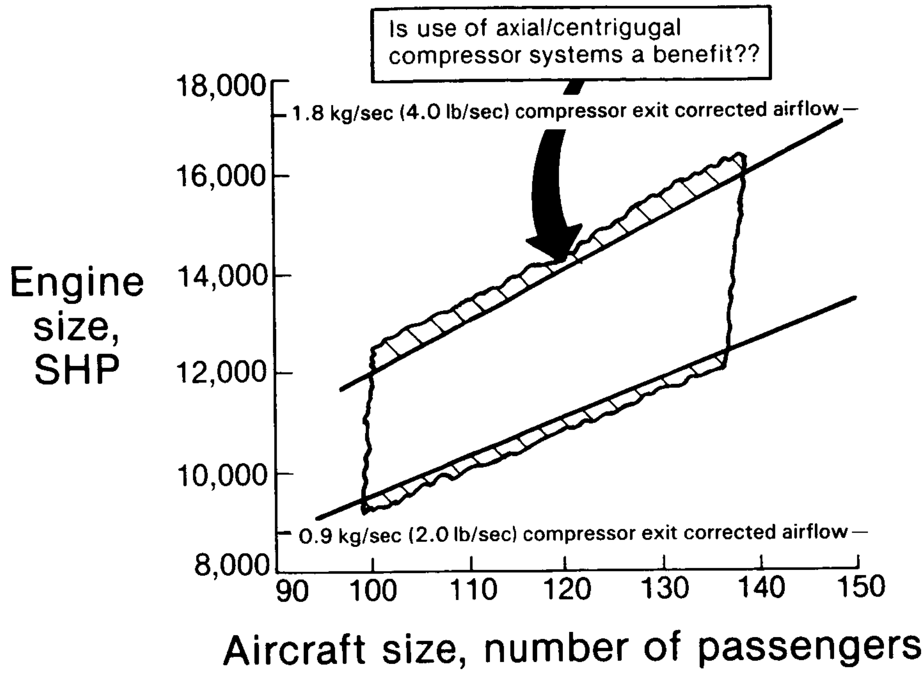

Figure 4.2-20 Evaluation of Axial and Axial/Centrifugal Compression Systems - Both axial and centrifugal compression systems are candidates for engines in 100 - 150 passenger aircraft. (J27638-56)

Engine Cycle - A reference cycle of 29:1 overall pressure ratio at the maximum climb point and 1426°C (2600°F) maximum takeoff combustor exit temperature was used to screen the four candidate engine configurations. Previous work performed by Pratt & Whitney indicated that modest increases in overall pressure ratio should not have a significant impact on the configuration comparison.

4.2.3.2 Screening Candidate Configurations

Using the reference cycle, the configuration study was initiated by defining the flowpaths for each of the four engines. The pressure ratio split between high and low spool and design characteristics of compressors, turbines and burners were then defined. Mechanical design effort established the rotor support scheme and included an assessment of critical speed (rotor dynamics). Engine fuel burn, direct operating cost, weight, price, and dimensions were determined and the engines ranked on the basis of these key parameters.

4.2.3.2.1 Mechanical Design and Aerothermodynamic Analysis

An interactive computer program was used to generate a flowpath for each engine configuration. Engine cross sections were then developed from these flowpaths. Characteristics of major components including the compressor, combustor, and turbine were specified. Key design analysis considerations such as materials, structural analysis and rotor dynamics were investigated.

Engine Configuration Flowpaths

The aerothermodynamic flowpaths for the two-spool axial, three-spool axial and three-spool axial/centrifugal configurations are presented in Figure 4.2-21. The reversed engine uses the same flowpath as the three-spool axial/ centrifugal engine configuration. These flowpaths were derived using consistent aerodynamic loading and material technology levels, producing the number of stages and axial and radial dimensions shown.

The flowpath for the three-spool all-axial configuration (STS647) is the longest of the three. The three-spool arrangement in this small size (10,000 horsepower) engine results in several mechanical and structural difficulties. The flowpath shown in the figure was found to be neither mechanically nor structurally feasible due to high bearing DN levels and inadequate space for the turbine disks. To resolve these problems, modifications to the high spool flowpath, including reduced rotor speed and increased diameter, were required.

The flowpath for the three-spool axial/centrifugal engine configuration (STS646) is the shortest of the three but the engine has the largest diameter. Use of a centrifugal compressor with folded burner provides room for an overhung high pressure turbine bearing arrangement, thus eliminating the need for an inter-turbine support strut.

The third flowpath, the all-axial two-spool configuration (STS648), eliminates the third concentric shaft by putting the low-pressure compressor on the same spool as the Prop-Fan.

Aerothermodynamic flowpath definitions were generated with an existing Pratt & Whitney analytical computer program. The engine flowpath definition forms the basis for adding the mechanical-structural features (i.e., disks, bearing compartments, shrouds, engine cases, etc.) resulting in an engine cross section. The flowpath program is also used to estimate engine weight, cost, and nacelle dimensions.

The major differences between the three flowpaths are summarized in Table 4.2-VII. The axial compression two-spool engine (STS648) incorporates a 12-stage compressor with a moderately high pressure ratio and a two-stage turbine. The low-pressure compressor is located on the same spool as the four-stage power turbine. The speed of the rotor is established by the close coupled turbine arrangement and the maximum turbine blade attachment stress (represented by AN^2). The three-stage compressor, which has a corrected tip speed of 353 m/sec (1160 ft/sec) provides a good balance between low-pressure compressor performance and engine weight and cost.

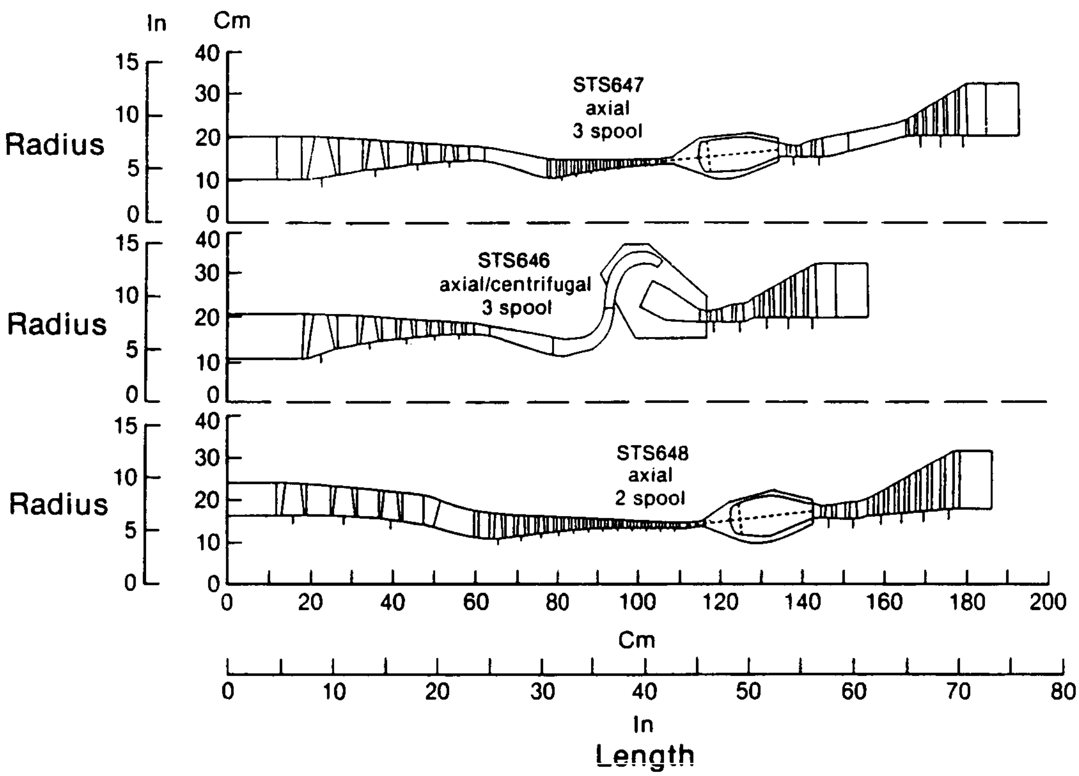

Figure 4.2-21 Engine Configuration Flowpaths - The three flowpaths were derived using 1988 technology for materials, cooling, and aerodynamics. The same flowpath is used for the three-spool axial/centrifugal engine and the reversed engine. (J27638-107)

The pressure ratio split in the three-spool axial/centrifugal engine (STS646) is 7.1 x 3.6, with a 3.6:1 pressure ratio in the single-stage centrifugal compressor. A single-stage turbine drives the centrifugal compressor, a single-stage intermediate turbine drives the five-stage low-pressure compressor, and a three-stage power turbine drives the Prop-Fan. In this configuration, high spool rotor speed is limited by the properties of the nickel-based centrifugal compressor disk. The speed of the intermediate spool is governed by optimum compressor efficiency. The speed of the power turbine is established by the close coupled turbine configuration and the maximum turbine blade attachment stress. This configuration draws heavily from the PW3005 engine designed by the Government Products Division of Pratt & Whitney.

The pressure ratio split in the three-spool axial compression engine (STS647) is 4.9 x 5.1. These pressure ratios were used to match the diameters of the high and intermediate turbine at the maximum allowable rotor speed. In this configuration, the speed of the high spool is limited by the close coupled turbine arrangement, the speed of the intermediate turbine is governed by compressor efficiency, and the speed of the low spool is set by performance, weight, and cost considerations.

TABLE 4.2-VIII
OVERALL FLOWPATH COMPARISON

	STS648	STS646	STS647
Engine Configuration	Axial Two-Spool	Axial/Centrifugal Three-Spool	Axial Three-Spool
Flight Condition	-----Aerodynamic Design Point-----		
Altitude/Mn	10,668 /0.75 (35,000 ft)	10,668 /0.75 (35,000 ft)	10,668 /0.75 (35,000 ft)
Compressors			
Pressure Ratio	2 x 12.5	7.1 x 3.6	4.9 x 5.1
Number of Stages	3 + 12	5 + 1	4 + 7
Corrected Tip Speed, m/sec (ft/sec)			
- Low Compressor	353 (1160)	441 (1450)	441 (1450)
- High Compressor	355 (1165)	271 (890)	334 (1130)
Turbines			
No. of Stages	2 + 4	1 + 1 + 3	1 + 1 + 3
AN^2 - Maximum			
- High Turbine	Base	-13%	Base
- Intermediate Turbine	--	Base	Base
- Power Turbine	Base	Base	Base
Overall Length	Base	-30 cm (-12 in)	+7 cm (+3 in)

Compressor - Using the aerothermodynamic flowpath definitions and data from previous studies, including the NASA-sponsored Energy Efficient Engine program, the characteristics of the high and low-pressure compressors were defined and evaluated.

Low-pressure compressor design parameters are compared in Table 4.2-IX. The low-pressure compressors in the study engines feature controlled diffusion airfoils, abradable tip rub strips, low aspect ratio, high loading, and low airfoil count. These and other advanced technology features are assumed to be available by 1988.

High-pressure compressor design parameters are compared in Table 4.2-X. The axial high-pressure compressors incorporate advanced technology features from the Energy Efficient Engine Program. Among these features are controlled diffusion airfoils, designed to provide low losses at high subsonic Mach numbers by controlling the diffusion process in the airfoil passage so that recompression is accomplished without shocks. Abradable blade tip rubstrips with trenches are used to reduce the sensitivity of efficiency to tip clearance. The rotor design features mini-cavities to reduce endwall losses. An active clearance control system is used for improved efficiency.

The centrifugal compressor draws on Pratt & Whitney of Canada design and service experience with centrifugal compressors.

TABLE 4.2-IX
COMPARISON OF MAJOR LOW-PRESSURE COMPRESSOR DESIGN PARAMETERS

Engine Configuration	STS648	STS646	STS647
	Axial Two-Spool	Axial/ Centrifugal Three-Spool	Axial Three-Spool
Pressure Ratio	2.0	7.1	4.9
Inlet Corrected Airflow, kg/sec (lb/sec)	18.1 (40.0)	18.8 (41.5)	18.3 (40.5)
Number of Stages	3	5	4
Corrected Tip Speed, m/sec (ft/sec)	353 (1160)	441 (1450)	441 (1450)
Inlet Hub/Tip Ratio	0.67	0.46	0.46
Aspect Ratio	1.2	1.2	1.2
Gap/Chord Ratio	0.6	0.6	0.6
Number of Airfoils	205	494	288
Length, cm (in)	35 (14)	43 (17)	40 (16)

TABLE 4.2-X
COMPARISON OF MAJOR HIGH PRESSURE COMPRESSOR DESIGN PARAMETERS

Engine Configuration	STS648	STS646	STS647
	Axial Two-Spool	Axial/ Centrifugal Three-Spool	Axial Three-Spool
Pressure Ratio	12.5	3.6	5.1
Inlet Corrected Airflow, kg/sec (lb/sec)	9.9 (22.0)	3.6 (8.0)	4.9 (11.0)
Number of Stages	12	1	7
Inlet Corrected Tip Speed, m/sec (ft/sec)	355 (1165)	271 (890)	344 (1130)
Hub/Tip Ratio			
- Inlet	.63	.70	.69
- Exit	.90	---	.90
Aspect Ratio	1.8	---	1.8
Gap/Chord Ratio	.93	---	.93
Number of Airfoils	1578	---	895
Length, cm (in)	48 (19)	22 (9)	25 (10)
Specific Speed	---	63	---

Combustor - The characteristics of the combustors used in the three configurations are compared in Table 4.2-XI. The axial compression engines use an advanced single stage aerating (MARK) burner. The overall pressure loss (diffuser plus liner) of the two-spool axial engine (STS648) is 3.2% of inlet pressure. The overall pressure loss of the three-spool (STS647) axial engine is 3.7%. The increase is due to mechanical considerations which require additional turning in the diffuser.

The three-spool axial/centrifugal engine (STS646) incorporates a single stage aerating burner which is canted at a 20 degree angle to mate with the centrifugal compressor. Pipe diffusers are used in this configuration: diffuser pressure loss is included in the centrifugal compressor efficiency. There is a 3% liner pressure loss in the combustor.

All three engines have the potential to meet the emissions requirements specified for the APET Program with developed combustion systems.

TABLE 4.2-XI
COMBUSTOR

	STS648	STS647	STS646
High Compressor Configuration	Axial	Axial	Centrifugal
Design Parameters			
Exit Temperature (CET), °C (°F)	1426 (2600)	1426 (2600)	1426 (2600)
Inlet Temperature (CIT), °C (°F)	511 (953)	503 (939)	522 (972)
Type of Combustor	Advanced Single Stage Aerating		Canted Single Stage Aerating
Pressure Loss, % P_t in			
Overall	3.2	3.7	---
Liner	2.0	2.0	3.0
Burning Length, cm	16.6/14.6	16.6/14.6	14.6
(in)	(6.5/5.75)	(6.5/5.75)	(5.75)
Number of Fuel Injectors	14	14	24
Space Heat Release Rate, Btu/hr ft^3 atm x 10^{-6}	5.9	5.9	5.6
Emissions	Potential for all to meet regulations		
Combustor Status			
Lean Blowout Fuel to Air Ratio	0.004	0.004	0.0095/.0066
Emissions, HC, CO, NOx	Test to be conducted		
Pattern Factor, $\frac{T_{max} - CET}{CET-CIT}$	0.20	0.20	0.16

Turbine Airfoil Cooling Requirements - Turbine airfoil cooling requirements for the three engine configurations are presented in Table 4.2-XII. The features assumed to be available for 1988 technology verification include: high cooling effectiveness attained by multipass, impingement showerhead, blade trailing edge discharge; use of advanced single crystal alloys with allowable metal temperatures 37°C (100°F) higher than current alloys; thermal barrier coatings on blades and vanes which increase allowable gas temperatures by 57°C (135°F); turbine inlet temperature profiles and pattern factors commensurate with the Energy Efficient Engine and PW3005 combustors.

TABLE 4.2-XII
TURBINE AIRFOIL COOLING REQUIREMENTS

	Maximum Combustor Exit Temperature = 1426°C (2600°F)		
	Two-Spool Axial	Three-Spool Axial/Centrifugal	Three-Spool Axial
1st Stage Vane	6.05%	5.8%	5.6%
Platform	0.30	0.25	0.25
1st Stage Blade	1.4	1.2	1.25
2nd Vane	0.5	0.5	0.5
2nd Blade	0.35	0.35	0.35
Total	8.6	8.1	7.95

Mechanical Description of the Candidate Engines

Mechanical design effort followed the flowpath and component definition. A mechanical cross section was prepared for each of the four candidate engine configurations.

Three-Spool Axial/Centrifugal Engine (STS646) - A cross section of the three-spool axial/centrifugal compression engine (STS646) is presented in Figure 4.2-22. In this engine, the front end of the power turbine shaft is supported by a large thrust bearing. With this arrangement, the axial blow-off loads imposed by the power turbine are taken out through the mechanical bearing, thus avoiding the performance penalty associated with injecting bleed air into the turbine rear cavity to thrust balance the blow-off load across the turbine.

The inlet case features struts with variable trailing edge flaps, used to optimize off-design performance operation. The intermediate case supports both the rear of the low compressor and the front of the high compressor. In addition, a drive shaft passes through the intermediate case which is used to start the engine and to drive engine accessories.

The high compressor features a centrifugal impeller. The air leaving the impeller is passed through pipe diffusers into the burner cavity. The burner section features the advanced technology segmented burner liner from the Energy Efficient Engine Program.

The bearing compartment between the impeller and the high turbine is similar to the bearing compartment used in the PW3005 engine. It features a buffered sealing system in which cooler, lower pressure, low compressor exit air surrounds the bearing compartment and protects it from the hot high compressor exit gas. Buffering air that leaks past the seals is reinjected back into the engine at the turbine exhaust region. This air does double duty; it is also used in the turbine exhaust inner cavity to provide additional thrust balance to minimize the blow-off load of the low turbine.

A piggyback bearing arrangement is used in the intermediate turbine. The intermediate shaft is supported by the power turbine shaft through an intershaft roller bearing: the power turbine shaft is in turn supported by another roller bearing to the turbine exhaust case. The piggyback bearing compartment features labyrinth seals in which low compressor discharge air is used to buffer and minimize the temperatures in the bearing compartment region. To minimize bearing case speed, the power turbine rotor and the intermediate turbine rotor are counter rotating.

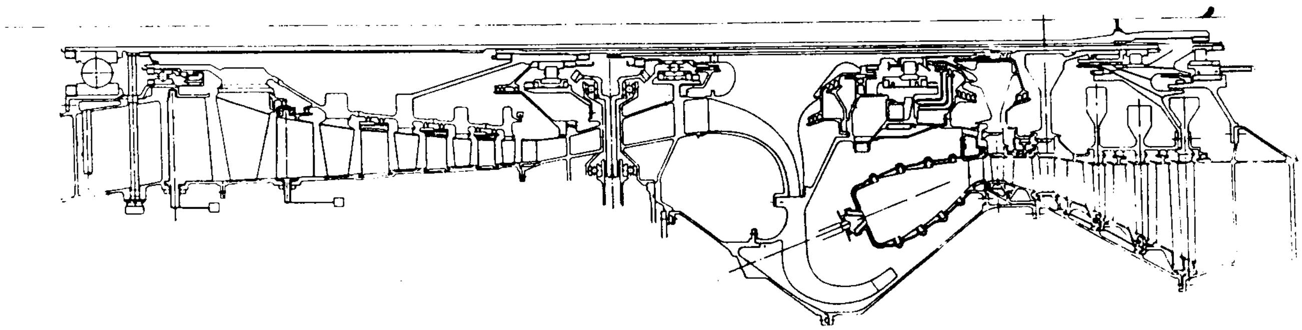

Figure 4.2-22 Mechanical Cross Section of the Three-Spool Engine with Axial/Centrifugal Compressor (STS646)- The major features include a large thrust bearing to support the power turbine shaft, inlet case struts with variable trailing edge flaps, a centrifugal impeller, and a piggyback bearing arrangement. (J27638-908)

Two-Spool Axial Engine (STS648) - The STS648 engine is a two-spool, all-axial compression configuration. A mechanical cross section of the engine is presented in Figure 4.2-23. In this configuration, the power turbine drives both the low compressor and the propeller drive shaft. The front end of the power turbine shaft is supported by a large thrust bearing, which also reacts the power turbine blow-off loads. This arrangement also minimizes the performance losses associated with using thrust balance bleed air to offset the blow-off load of the power turbine. The intermediate case supports both the rear of the low compressor and the front of the high compressor. The starter drive shaft, which passes through the intermediate case, is also used to drive engine accessories. The high compressor features four stages of variable geometry in the front end and a compact inner stator seal arrangement aimed at reducing the losses associated with inner seal cavity airflow leakage.

An advanced single stage burner is used in the combustor. The burner incorporates the segmented cooling liner currently under evaluation and technology development in the Energy Efficient Engine Program.

The rear of the high spool is supported from the power drive shaft by a piggyback support system. This piggyback bearing system is very similar to the configuration used in the three-spool axial/centrifugal engine (STS646). In the two-spool engine, compressor bleed air is also a source of low temperature buffer air used to control temperatures in the bearing compartment.

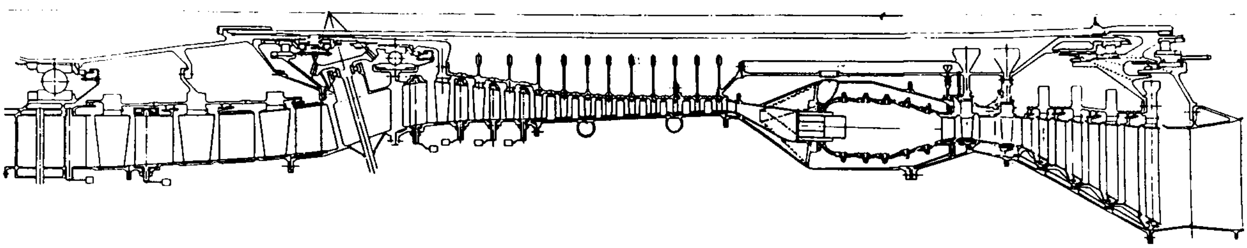

Figure 4.2-23 Mechanical Cross Section of the Two-Spool All-Axial Engine (STS648) - The power turbine drives both the low compressor and the propeller. (J27638-909)

Three-Spool Axial Engine (STS647) - The STS647 engine is a three-spool axial compressor configuration. A cross section of the engine is presented in Figure 4.2-24. Schematically the engine has many of the features used in the two-spool axial and three-spool axial/centrifugal configurations (STS646 and STS648). The major difference in the design of the three-spool axial engine is the use of a bearing turbine intermediate case support strut system. This arrangement was selected because the high rotor is more easily supported with a strut than a bearing in the cramped space underneath the burner.

During the conceptual design phase it was determined that the speed selected to define the high rotor flowpath, combined with the mechanical requirement of having three shafts pass through the bore region of the high turbine disk, resulted in an extremely difficult design problem; providing adequate disk structure in the small amount of space available. A larger diameter, lower speed high spool would be required to allow for a properly designed high pressure turbine disk with adequate strength. Another result of the speed originally chosen for the high rotor was a high bearing DN level for the high rotor support system. While the conceptual engine configuration and flowpath were not modified, it was estimated that the high spool flowpath would have to be moved out in diameter approximately 5 cm (2 in) to alleviate the problems noted. The impact of this change on engine performance, weight, acquisition cost, maintenance cost, and bearing speeds has been accounted for in the final evaluation of this engine configuration. In spite of these modifications, the three-spool axial compression engine still has the highest bearing DN levels of any configuration evaluated.

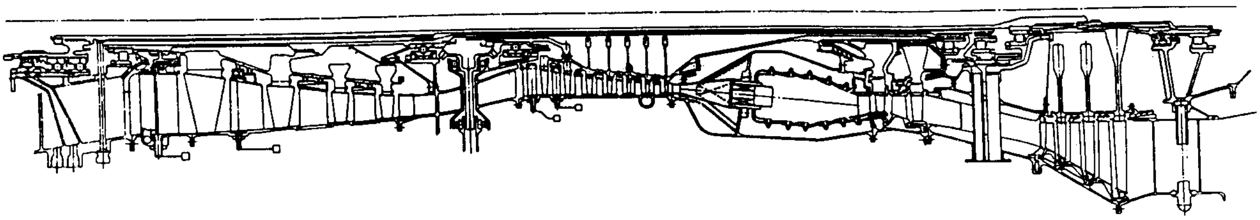

Figure 4.2-24 Mechanical Cross Section of the Three-Spool All-Axial Engine (STS647) - The major feature of this engine is a hot strut bearing support system for the high rotor. (J27638-910)

Reversed Three-Spool Engine (STS646R) - The STS646R engine is a reversed three-spool axial/centrifugal compression configuration. The major objective of this novel design was to eliminate the third concentric shaft in the engine. A cross section of the engine is presented in Figure 4.2-25. In this reversed configuration, the turbine is located at the front of the engine and the compressor at the rear.

Most of the features of this engine are similar to the three-spool axial/centrifugal engine, STS646. However, the reversed configuration includes a support strut which supports the back end of the power turbine.

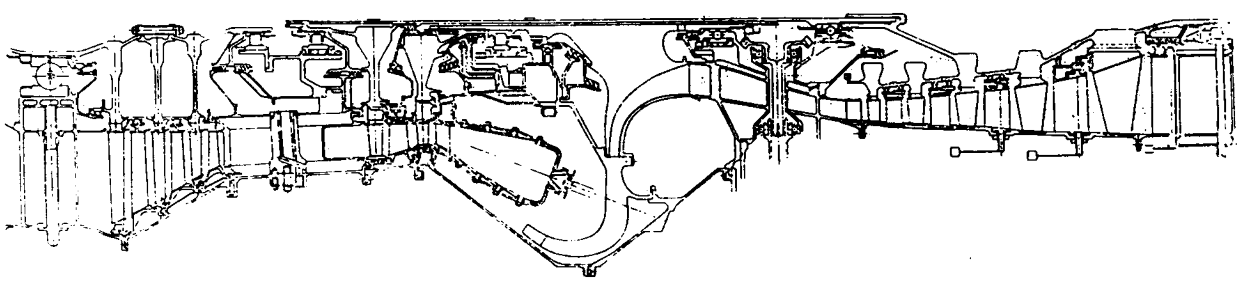

Figure 4.2-25 Mechanical Cross Section of the Reversed Three-Spool Axial/Centrifugal Engine - In this reversed configuration, the turbine is located at the front and the compressor at the rear. A turbine intermediate case bearing support strut system is used to support the back end of the power drive turbine. (J27638-911)

Structural Analysis, Materials, and Rotor Dynamics

As mechanical cross sections were being developed, critical issues related to engine structure, materials, and rotor dynamics were addressed.

Critical Structural Analysis Considerations - Structural design limits were identified for the three study engines; critical considerations are covered in Table 4.2-XIII. In most cases, design limits are set by blade roots and attachments. However, bore stresses are the limiting factor in the centrifugal compressor.

TABLE 4.2-XIII
CRITICAL STRUCTURAL ANALYSIS CONSIDERATIONS

STS648 (Two-Spool Axial)	
o Power Turbine Spool	Allowable stress in last stage turbine blade/attachment sets design limits
o High Spool	Allowable stress in second stage turbine blade/attachment sets rotor speed limit
STS646 (Three-Spool Axial/Centrifugal)	
o Power Turbine Spool	Allowable stress in last stage turbine blade/attachment sets rotor speed limit
o Intermediate Spool	Allowable stress in turbine blade attachment and compressor aerodynamic/LCF life considerations set rotor speed
o High Spool	LCF life/centrifugal disk bore stresses set rotor speed
STS647 (Three-Spool Axial)	
o Power Turbine Spool	Allowable stress in last stage turbine blade/attachment sets rotor speed
o Intermediate Spool	Allowable stress in turbine blade/attachment and compressor aerodynamic (LCF) life considerations set rotor speed
o High Spool	Allowable stress in turbine blade/attachment and disk bore allowable stress (LCF life) set rotor speed

Materials - The materials considered for use in the study engines are specified in Table 4.2-XIV. Advanced aluminum will be used for components operating at temperatures up to 260°C (500°F). High fatigue strength is required for blades; therefore, titanium and advanced nickel alloys will be used as temperatures increase. High tensile strength and weldability are important for individual disks or drum rotors. For these components, material selection will also be determined by operating environment. Inconnel 100 nickel alloy would be used for the centrifugal compressor. Second generation single crystal alloys with thermal barrier coatings would be used for the cooled turbine airfoils.

TABLE 4.2-XIV
MATERIALS

Component	Material
Low Pressure Compressor	
Blades	Advanced Al
Disks (Drum)	Advanced Al
Vanes	Al Cast to Size
Cases	Advanced Al
Intermediate Case	17-4PH Steel
High Pressure Compressor	
Blades - to 260°C	Advanced Al
Blades - 260°C to 482°C	Ti
Blades - Above 482°C	Ni Alloy
Vanes - to 260°C	Advanced Al
Vanes - 260°C to 482°C	Ti
Vanes - Above 482°C	Ni Alloy
Disks - to 260°C	Advanced Al
Disks - 260°C to 482°C	Ti
Disks - Above 482°C	Ni Alloy
Cases - to 482°C	Steel
Cases - above 482°C	Ni Alloy
Centrifugal Compressor	IN100
Burner	
Liner	B1900
Outer Liner	B1900
Cases	Ni Alloy
Turbine	
First and Second Stage Blades	Second generation single crystal Ni alloy and thermal barrier coating
First Stage Vanes	Second generation single crystal Ni alloy and thermal barrier coating
Second Stage Vanes	Triaxial, otherwise same as first stage vanes
First and Second Stage Disks	Advanced IN100
Power Turbine Blades	
Above 648°C	IN713
Below 648°C	Al-Ti
Disks	
Above 537°C	IN100
Below 537°C	Ti
Cases	Ni Alloy
Shafts	AMS6304

Rotor Dynamics - A rotor dynamics evaluation relative to design criteria for the three major configurations is shown in Table 4.2-XV. Initially, none of the engines met the required power turbine idle margin. However, the margin was met with a combination of shaft redesign and increased idle speed.

TABLE 4.2-XV
ROTOR DYNAMICS

Criteria: Critical Speed must be
15% Below Idle
or 15% Above Red Line

	Idle % Margin	Red Line % Margin
STS646 (3-Spool Axial/Centrifugal)		
Power Turbine Spool	-15	+100
Intermediate Spool	-21	+ 21
High Spool	Sufficient Margin	
STS647 (3-Spool Axial)		
Power Turbine Spool		
Intermediate Spool	Similar to STS646	
High Spool		
STS648 (2-Spool Axial)		
Turbine Spool	-15	+ 30
High Spool	Sufficient Margin	

Bearings, Leakage, and Thrust Balance - Mechanical design effort was completed with analysis of bearings, leakage flow, and thrust balance. Although some changes would have to be made in the final design, none of the configurations was eliminated from consideration due to bearing, leakage, or thrust balance technical issues.

Bearing DN's are shown in Table 4.2-XVI. These levels represent moderate advances over current technology.

TABLE 4.2-XVI
TURBOPROP ENGINE BEARINGS

	Bearing DN Levels (Millions)		
	Two-Spool Axial (STS648)	Three-Spool Axial/Centrifugal (STS646)	Three-Spool Axial (STS647)
Power Turbine Thrust	2.0	1.7	1.7
Low Rotor Thrust	2.0	2.0	2.1
Low Rotor Forward Roller	1.0	2.1	2.0
High Rotor Thrust	2.3	2.5	2.5
High Rotor Roller	1.9*	2.5	2.5*
Low Rotor Roller	1.0*	1.6*	2.1*
Low Rotor Rear	0.9	---	2.1
Power Turbine Roller	---	1.0*	0.9
Power Turbine Rear	---	1.0	1.3

*Piggyback bearings

Secondary flows and thrust balances for the three configurations are shown in Table 4.2-XVII. In setting secondary cooling and leakage flows, it is assumed that advances will be made in static sealing and cooling air delivery system technology.

A preliminary study indicated that there are no major differences in secondary flow between the two-spool and three-spool engines. However, additional seal work, and particularly a low leakage, high speed seal for piggyback bearings, is required to achieve the projected flow levels. The large diameter of the power turbine thrust bearing in the two-spool configuration is likely to provide an advantage in low rotor thrust balance over the other engines.

TABLE 4.2-XVII
SECONDARY FLOWS AND THRUST BALANCE

	Two-Spool Axial	Three-Spool Axial/ Centrifugal	Three-Spool Axial
Airfoil Cooling, Percent	8.6	8.1	8.0
Secondary Cooling and Leakage, Percent	4.2	4.2	4.2
Total Cooling and Leakage, Percent	12.8	12.3	12.2

The conclusions base on these analyses are:

- o No secondary flow difference between twoand three-spool engines
- o Large power turbine thrust bearing used to eliminate thrust balance leakage airflow penalty
- o High speed, low leakage seals required for piggyback bearings

4.2.3.2.2 Performance Related Issues

After the mechanical design effort was completed, the performance of the four candidate turboprop engines was evaluated. The analysis covered engine fuel burn and component efficiency at the aerodynamic design point, the impact of deterioration on engine performance, the effects of customer bleed and gearbox power extraction, part power performance, operational constraints and starting requirements. Three of the engines remained competitive, but the reversed engine exhibited unacceptable inlet and exhaust system pressure losses, resulting in a significant disadvantage in fuel burn relative to the other concepts evaluated.

Ground Rules for Performance Evaluation

A detailed set of ground rules was developed for the engine performance evaluation (see Table 4.2-XVIII). Use of these ground rules ensured a consistent, objective evaluation of the performance characteristics of the engine configurations.

Holding thrust ratio and climb/cruise thrust margin constant ensured that each engine would be rated consistently at critical operating conditions. Thrust levels were tailored to the requirements of the 120-passenger reference aircraft.

Maintaining constant propeller tip speed and power loading limited the impact of the Prop-Fan on engine performance. Previous work indicated that a tip speed of 243 m/sec (800 ft/sec) and power loading of 34 shp/D^2 resulted in an efficient Prop-Fan operating condition.

Primary stream jet velocity was held constant for the performance evaluation. A follow-on study of power turbine work extraction indicated that a velocity of 304 m/sec (1000 ft/sec) at the aerodynamic design point did provide the best combination of fuel burn and direct operating cost.

Selecting a consistent growth philosophy aided in the objective evaluation of engine performance. Ten percent growth in shaft horsepower reflects projected future applications for Prop-Fan propulsion.

TABLE 4.2-XVIII
PERFORMANCE RELATED ISSUES
(Ground Rules for Configuration Study)

o Constant Climb-Thrust/Takeoff-Thrust Ratio

$$\frac{\text{Fn } 10{,}668 \text{ m } (35{,}000 \text{ ft}),\ 0.75\text{M Std.}}{\text{Fn SL } 0.22\text{M } +14^\circ\text{C } (+25^\circ\text{F})} = 0.24$$

o Constant Climb-Thrust/Cruise-Thrust Ratio

$$\frac{\text{Fn MCL } 10{,}668 \text{ m } (35{,}000 \text{ ft}),\ 0.75\text{M}}{\text{Fn MCR } 10{,}668 \text{ m } (35{,}000 \text{ ft}),\ 0.75\text{M}} = 1.09$$

o Propeller Loading (shp/D^2) @ Maximum Climb (10,668 m (35,000 ft), 0.75M) = 34 hp/ft^2, U_t = 243 m/sec (800 ft/sec)

o Constant Speed Operation of Propeller During Takeoff, Climb and Cruise

o Constant Power Turbine Work Extraction Philosophy (V_{je} = 304 m/sec (1000 ft/sec))

o 10% Future Shaft Horsepower Growth Considered for all Configurations

General Characteristics of the Engine Configurations

Table 4.2-XIX shows the general operating parameters for the engine configurations at the aerodynamic design point, climb, cruise, and takeoff conditions. The cycle pressure ratio used in the configuration selection study (Table 4.2-XIX) is lower than the pressure ratio derived from the cycle optimization (Table 4.2-VI). Since the two studies were conducted in parallel, the optimum pressure ratio was not known when the configuration study was initiated.

TABLE 4.2-XIX
SUMMARY OF GENERAL CHARACTERISTICS FOR ALL ENGINE CONFIGURATIONS

Propeller Diameter, m (ft)	3.5 (11.5)
Aerodynamic Design Point at 10,668 m (35,000 ft), M = 0.75	
Overall Pressure Ratio	25
Shaft Horsepower, hp	3870
Shp/D^2, hp/ft^2	29.2
Thrust, N (lb)	10,408 (2340)
Maximum Climb at 10,668 m (35,000 ft), M = 0.75	
Overall Pressure Ratio	29
Shaft Horsepower, hp	4500
Shp/D^2, hp/ft^2	34
Thrust, N (lb)	13,344 (3000)
Maximum Cruise at 10,668 m (35,000 ft), M = 0.75	
Overall Pressure Ratio	27
Shaft Horsepower, hp	4115
Shp/D^2, hp/ft^2	31
Thrust, N (lb)	12,232 (2750)
Maximum Takeoff Sea Level Static +14°C (+25°F)	
Shaft Horsepower, hp	9230
Shp/D^2, hp/ft^2	69.8
Thrust, N (lb)	68,947 (15500)
Combustor Exit Temperature, °C (°F)	1426 (2600)
Overall Pressure Ratio	23.2

Component Performance at Maximum Cruise

The performance characteristics of the major components in the three engine configurations are compared in Table 4.2-XX.

TABLE 4.2-XX
COMPONENT PERFORMANCE COMPARISON AT MAXIMUM CRUISE RATING
(10,668 m (35,000 ft), M = 0.75)
(109 Horsepower Taken from Gearbox)

	STS648 Two-Spool All Axial	STS647 Three-Spool All Axial	STS646 Three-Spool Axial/Centrifugal
Low Compressor			
Pressure Ratio	2.1	5.3	7.7
Polytropic Efficiency, %	Base	+1.5	+0.7
High Compressor			
Pressure Ratio	13	5.2	3.6
Polytropic Efficiency, %	Base	-0.8	-2.6
Overall Compression System			
Polytropic Efficiency, %	Base	+0.3	-0.6
Combustor			
Pressure Loss, %	Base	+0.5	+1.1
High Pressure Turbine			
Expansion Ratio	3.49	2.5	2.1
Efficiency, %	Base	-2.2	-1.9
Low Turbine			
Expansion Ratio	----	1.8	2.1
Efficiency, %	----	Base	+1.7
Power Turbine			
Expansion Ratio	7.9		
Efficiency, %	Base	+0.2	+0.1

New Engine Performance at Maximum Cruise

The performance of the three configurations at the maximum cruise rating is compared in Table 4.2-XXI. As indicated previously (page 77), the overall pressure ratio is lower than the pressure ratio for the optimum cycle. The comparison presented in the table is based on specific fuel consumption for a new engine. It is assumed that the engines incorporate technology features appropriate for commercial certification in 1992.

C-2

The best performance is obtained with the two-spool, all-axial configuration. However, the differences in the performance of the three-spool all-axial configuration (0.4% poorer thrust specific fuel consumption) and the three-spool axial/centrifugal configuration (0.3% poorer TSFC) are not considered significant.

TABLE 4.2-XXI
ENGINE PERFORMANCE COMPARISON AT MAXIMUM CRUISE RATING
(10,668 m (35,000 ft), M = 0.75)
(109 Horsepower Taken from Gearbox)

	STS648 Two-Spool All Axial	STS647 Three-Spool All Axial	STS646 Three-Spool Axial/Centrifugal
Overall Pressure Ratio	27	27.3	27.5
Combustor Exit Temperature, °C (°F)	1273 (2325)	1261 (2303)	1260 (2300)
Shaft Horsepower, hp	4000	4000	4000
Difference in BSFC	Base	+0.4%	+0.3%
Total Thrust, N (lb)	11,832 (2660)	11,832 (2660)	11,832 (2660)
Difference in TSFC	Base	+0.4%	+0.3%

Engine Performance Deterioration

Performance deterioration was evaluated by identifying the factors which cause the performance of axial and centrifugal engines to deteriorate and then assessing the impact of these factors on the thrust specific fuel consumption of the study engines. Since the major differences in the deteriorated performance of axial and centrifugal engines result from variations in compressor efficiency, compressor performance characteristics were the focal point of this evaluation.

Effects of Performance Deterioration - Table 4.2-XXII highlights Pratt & Whitney experience in evaluating performance deterioration in axial and centrifugal compression engines.

Effect of Deterioration on Performance Comparison - The effects of deterioration on engine performance are shown in Table 4.2-XXIII. Pratt & Whitney of Canada experience indicated that the performance of centrifugal compressors deteriorated at a much slower rate than the performance of axial compressors. Therefore, the three-spool axial/centrifugal engine exhibits better thrust specific fuel consumption after 3500 cycles than either of the axial configurations.

The slightly poorer thrust specific fuel consumption of the two-spool axial compression engine after 3500 cycles is not considered a significant disadvantage. However, the thrust specific fuel consumption of the three-spool axial compression engine after 3500 cycles is considerably worse than the TSFC of the three-spool axial/centrifugal engine.

TABLE 4.2-XXII
EFFECTS OF PERFORMANCE DETERIORATION
(10,000 shp, 0.9 kg/sec (2 lb/sec) Exit Flow)

Axial Compression Engines

- Axial compressor model based on extensive JT8D, JT9D airline data
- Axial compressors in study engines STS647 and STS648 are likely to experience similar types of damage:

  Tight clearances
  Rubstrips
  Engine inlet ground clearance

- Polytropic efficiency of axial compressors is assumed to deteriorate about 2.5% in 3500 cycles because of small size airfoils

Centrifugal Compression Engines

- Centrifugal compressor model based on Pratt & Whitney of Canada experience
- Based on Pratt & Whitney of Canada experience, axial compressor blades are replaced three times more frequently than centrifugal
- Polytropic efficiency of centrifugal compressors is assumed to deteriorate 0.7% in 3500 cycles

TABLE 4.2-XXIII
EFFECT OF DETERIORATION ON PERFORMANCE COMPARISON

	STS648 Two-Spool All-Axial	STS647 Three-Spool All-Axial	STS646 Three-Spool Axial/Centrifugal
New Engine Performance			
Difference in TSFC	Base	+0.4%	+0.3%
Deteriorated Performance After 3500 Flight Cycles			
Difference in TSFC	Base	+0.4%	-0.3%

Reversed Engine Configuration - STS646R

The installed configuration concept and performance characteristics of the "reversed" three-spool axial/centrifugal engine, STS646R, are shown in Figure 4.2-26. Reversing the conventional three-spool axial/centrifugal engine, STS646, creates the requirement to turn inlet flow and exhaust flow as shown in the figure.

The inlet and exhaust system pressure losses resulting from the reversed flow are 7.5% and 5%, respectively. These losses result in a 4.2% increase in thrust specific fuel consumption relative to the conventional STS646 engine.

In addition, the external pod drag, which was not evaluated, was judged to be greater than the pod drag of the STS646, STS647, or STS648 because two exhaust pipes are exposed to a M = 0.75 flow field. The probability that a longer nacelle will be required also impacts the performance of the reversed engine configuration.

When all of these factors are considered, the reversed engine concept suffers a significant disadvantage relative to the conventional axial and axial/centrifugal configurations.

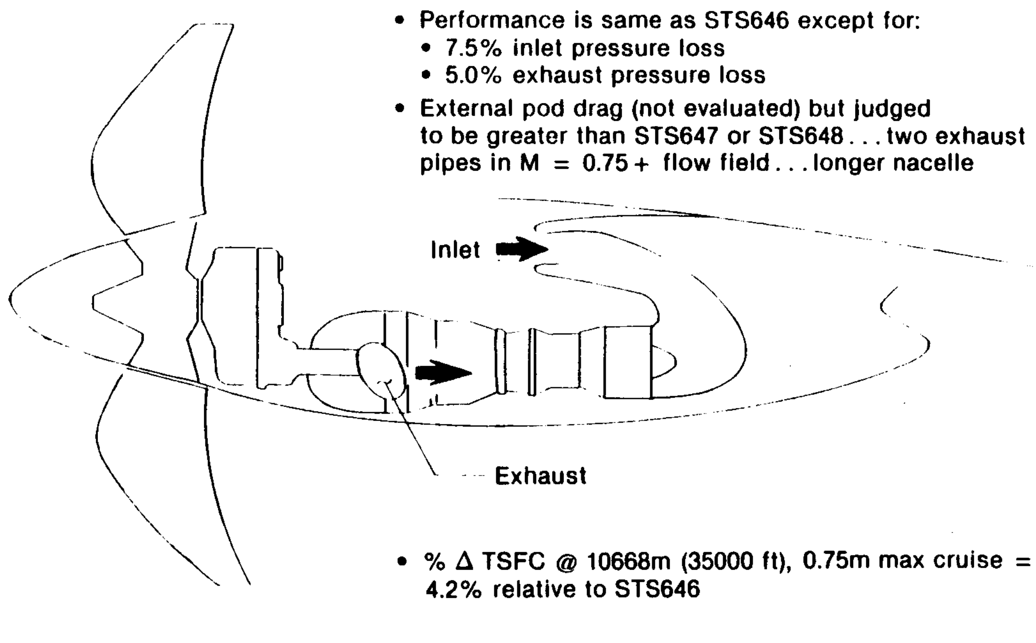

Figure 4.2-26 Reversed Engine Performance Characteristics - Inlet and exhaust system pressure losses encountered with the reversed engine led to a significant disadvantage in thrust specific fuel consumption. (J27638-57)

Effect of Customer Bleed on Performance

The impact of extracting 0.2 kg/sec (0.5 lb/sec) customer bleed from the high compressor discharge of the four engine configurations is shown in Table 4.2-XXIV. A similar increase in thrust specific fuel consumption is observed in all four engines at both the maximum cruise and 50% maximum cruise conditions.

The small airflow two-spool axial compression engine (STS648) experiences the greatest thrust loss. At 50% maximum cruise, turbine temperatures are significantly below rated levels; therefore thrust loss can be recovered by advancing the throttle setting. The effects of bleed on the performance of all four engines is essentially the same.

TABLE 4.2-XXIV
EFFECTS OF CUSTOMER BLEED ON PERFORMANCE
(0.2 kg/sec (0.5 lb/sec) High Compressor Discharge Bleed
10,668 m (35,000 ft) M = 0.75)

	STS648 Two-Spool All-Axial	STS647 Three-Spool All-Axial	STS646 Three-Spool Axial/ Centrifugal	STS646R (Reversed 646)
Maximum Cruise Rating				
Thrust Loss	11.5%	10.6%	10.5%	10.5%
TSFC Increase	+4.9%	4.8%	4.8%	4.8%
50% Maximum Cruise				
TSFC Increase	5.7%	5.6%	5.6%	5.6%

Effect of Gearbox Power Extraction on Performance

The effect of gearbox power extraction on engine performance at various power settings is shown in Table 4.2-XXV. As illustrated, the impact of horsepower extraction on all four engines is identical at a given power setting.

Part Power Performance

The relative part power performance characteristics of the two-spool and three-spool engine configurations are shown in Figure 4.2-27. The thrust specific fuel consumption of the two-spool engine is slightly higher than the TSFC of the three-spool engines between 50% and 100% maximum cruise. This differ- ence is due to the variable stator vanes used in the low-pressure compressor of the two-spool engine.

TABLE 4.2-XXV
EFFECT OF GEARBOX POWER EXTRACTION ON PERFORMANCE
(109 Horsepower Taken from Gearbox
10,668 m (35,000 ft) M = 0.75)

	STS648 Two-Spool All Axial	STS647 Three-Spool All Axial	STS646 Three-Spool Axial/ Centrifugal	STS646R (Reversed 646)
Maximum Cruise Rating				
Thrust Reduction	2.4%	2.4%	2.4%	2.4%
TSFC Increase	2.4%	2.4%	2.4%	2.4%
50% Maximum Cruise Thrust				
TSFC Increase	4.0%	4.0%	4.0%	4.0%

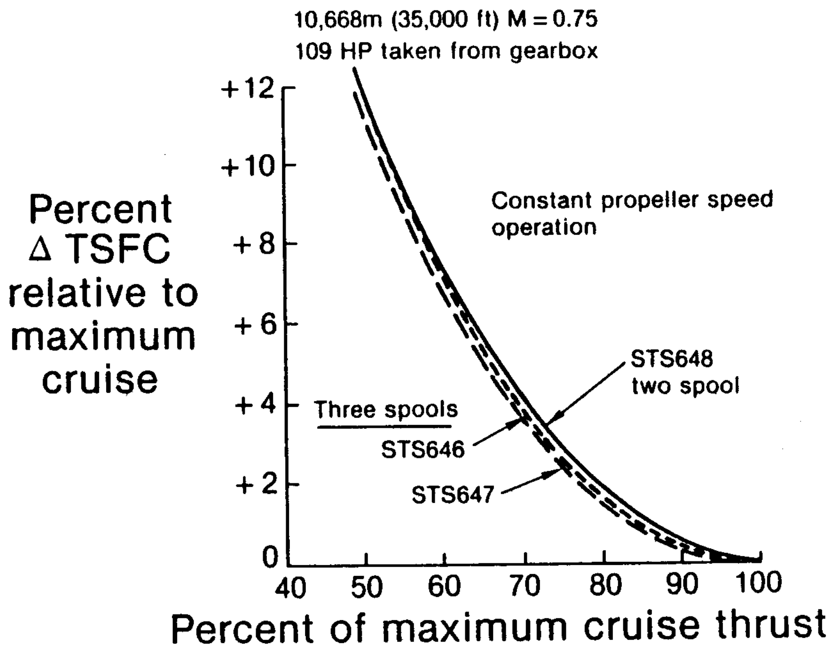

Figure 4.2-27 Off-Design Engine Matching Impact of Part Power Performance - The three-spool engines have slightly lower thrust specific fuel consumption at part power than the two-spool engine. (J27638-58)

Operational Constraints (Propeller Speed Schedule)

The propeller speed schedules for the two-spool and three-spool engine configurations are shown in Figure 4.2-28. The two-spool axial engine (STS648) provides constant speed over a range of power settings from 50% maximum cruise to approximately 110% maximum cruise. However, below about 50% maximum cruise the propeller must operate at variable speed. In contrast, the three-spool engines (STS646 and STS647) could operate at constant speed over the entire range of power settings.

If the propeller is not required to operate at constant speed, the two-spool engine will operate along the schedule shown in the lower portion of the figure. Both of the three-spool engines can operate at a specified constant speed or along a speed schedule.

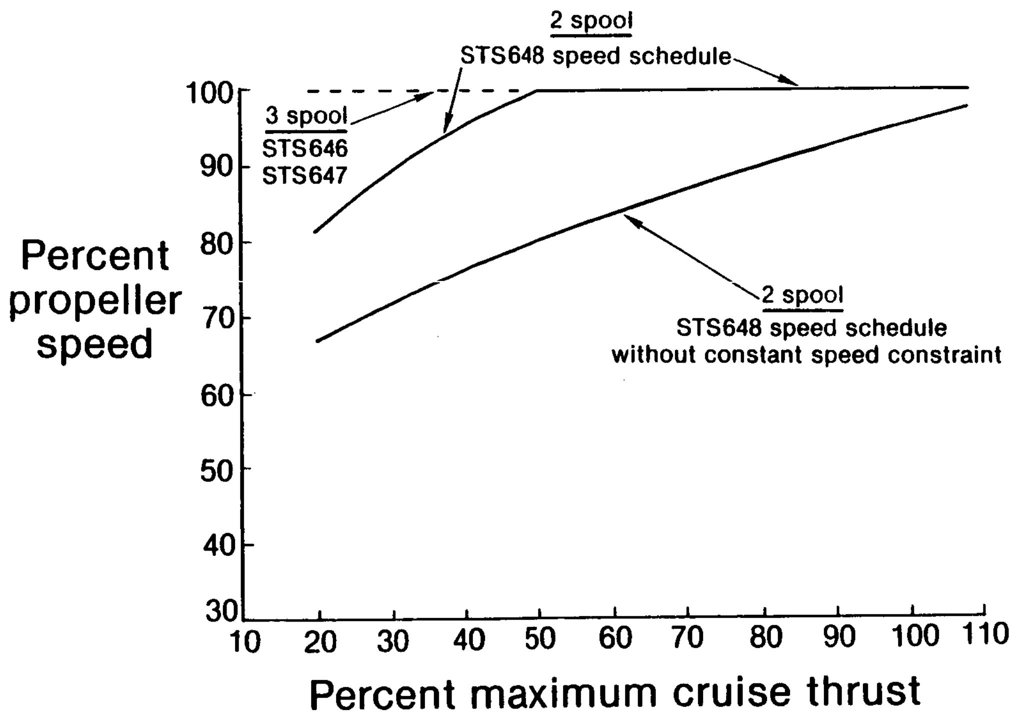

Figure 4.2-28 Operational Constraints - The three-spool engines provide a more flexible propeller speed schedule than the two-spool engine. (J27638-59)

Starting Requirements

The starter horsepower requirement for the two-spool and three-spool configurations are shown in Table 4.2-XXVI. The three-spool configurations, which have a lower high compressor pressure ratio, require the least starter horsepower.

TABLE 4.2-XXVI
STARTING REQUIREMENTS

	STS648 Two-Spool All-Axial	STS647 Three-Spool All-Axial	STS646 Three Spool Axial/Centrifugal
Takeoff Airflow	39	39	40
HPC Pressure Ratio	11.86	5.0	3.5
Relative Starter HP Required	Base	-45%	-55%

Propeller Drag in Connection with Failure Modes

A judgmental rating of free rotor windmilling drag is presented in Table 4.2-XXVII. The windmilling drag of the two-spool engine was judged to be slightly worse than the drag of either of the free turbine configurations because of the higher power required in the low rotor of the two-spool engine because the low-pressure compressor is on the low shaft. Since the increased power must ultimately come from the propeller in the windmilling mode, this was judged to result in slightly more drag for the two-spool configuration.

TABLE 4.2-XXVII
PROPELLER DRAG IN CONNECTION WITH FAILURE MODES

	STS648 Two-Spool All Axial	STS647 Three-Spool All Axial	STS646 Three Spool Axial/Centrifugal
Windmilling Drag	Base	Slightly Lower	Slightly Lower

4.2.3.2.3 Design Assurance Related Issues

The critical issues in the design assurance analysis are engine weight, acquisition cost, reliability, and maintenance cost. The four engine configurations were evaluated with the results summarized in Table 4.2-XXVIII.

For the most part, differences in engine weight, acquisition cost, and maintenance cost can be attributed to differences in the high-pressure compressor configuration (axial vs centrifugal compressor, 7-stage vs 12-stage axial compressor, etc.). Differences in compressor weight are influenced by the mass of the centrifugal disk compared to the number of axial stages.

In this study, weight estimates were made by a component analysis estimating technique in which key features of components were evaluated by comparability analysis which accounts for new technology and operating conditions.

Acquisition cost was determined using a program which analyzes key features of an engine flowpath such as number of stages, airfoil quantity and size, and flowpath diameter. These parameters were compared to a base engine with established material and labor costs. Unusual features, design concepts, and materials were accounted for.

Differences in the acquisition cost of the axial engines can be attributed to differences in the number of stages. The acquisition costs of the axial/centrifugal engines are generally lower because they contain fewer major parts.

Maintenance costs reflect both acquisition cost and reliability factors. Differences in engine reliability are inversely related to the number of bearings, number of intershaft bearings, number of major structures, and number of variable geometry vane stages.

Maintenance cost was evaluated by comparing the design features of a study engine to a base design, analyzing hot section lives and, thus determining reliability by mission and operational severity factors. Key design and operational parameters were compared to an existing engine with an established reliability and maintenance cost base. New reliability and maintenance cost estimates were generated by integrating detailed part lives, task man-hours, and module repair rates with price estimates consistent with acquisition cost. The effect of new design concepts and materials was also integrated into the model.

4.2.3.2.4 Environmental Issues

The noise and emissions characteristics of the four engine configurations were evaluated and found to be approximately equal. All four propulsion systems are capable of achieving the noise levels specified in Federal Aviation Regulation Part 36. Although some development work is required, all four configurations also have the potential to meet the International Civil Aviation Organization emissions goals, used as the standard for the APET Program.

4.2.3.2.5 Summary of Results

In the screening process, two engine configurations emerged as promising powerplants for the reference aircraft: the two-spool axial compression engine (STS648) and the three-spool axial/centrifugal compression engine (STS646). These two engines were considered approximately equal when fuel burn, direct operating cost, technical risk, and other performance, design, and environmental issues were integrated. The two configurations were then updated to reflect the optimum cycle and reevaluated to determine which configuration was best suited for engine/airplane integration studies with the airframe manufacturers. Results of the evaluation with the updated cycle are presented in the following section.

The characteristics of the four candidate engine configurations are summarized in Table 4.2-XXIX. After analyzing the results of the screening study, the two-spool axial and three-spool axial/centrifugal configurations were selected for further evaluation. The performance of the three-spool axial compression engine was inferior to the performance of the two-spool axial compression configuration, while the direct operating cost of the three-spool axial compression engine was inferior to the operating cost of the three-spool axial/centrifugal compression engine. Poor fuel burn characteristics caused the reversed engine configuration to be eliminated from further consideration.

TABLE 4.2-XXVIII
DESIGN ASSURANCE RELATED ISSUES

Engine Configuration	STS648 Axial 2-Spool	STS646 Axial/Centrifugal 3-Spool	STS647 Axial 3-Spool	STS646R Reversed Axial/ Centrifugal 3-Spool
Compressor Pressure Ratio	2 x 12.5	7.1 x 3.5	4.9 x 5.1	7.1 x 3.5
Compressor Stages	3 + 12	5 + 1	4 + 7	5 + 1
Turbine Stages	2 + 4	1 + 1 + 3	1 + 1 + 3	1 + 1 + 3
Number of Bearings	5	7	8	7
Maximum Bearing DN	2.3	2.5	2.5	2.5
Bearing Supports and Major Structures	3	4	4	5
Rotor Support System	Piggyback Bearings	Overhung Turbine/ Piggyback Bearings	Piggyback Bearings/ Hot Strut	Overhung Turbine/ Hot Strut
Overall Length, cm (in)	Base	-30 (-12)	+8 (+3)	-15 (-6)
Weight	Base	+7%	+13%	+13%
Acquisition Cost	Base	-7%	-1%	-4%
Reliability	Base	Base	Lowest	Lower
Maintenance Considerations				
Number of Modules	6	5	9	7
Maintenance Cost	Base	-16%	-6%	-15%

TABLE 4.2-XXIX
CONFIGURATION EVALUATION SUMMARY

	STS648 Two-Spool Axial	STS646 Three-Spool Axial/ Centrifugal	STS647 Three-Spool Axial	STS646R Reversed Three-Spool Axial/ Centrifugal
Compressor Stages	3 + 12	5 + 1	4 + 7	5 + 1
Turbine Stages	2 + 4	1 + 1 + 3	1 + 1 + 3	1 + 1 + 3
Number of Bearings	5	7	8	7
Maximum Bearing DN (millions)	2.3	2.4	2.5	2.4
Bearing Supports and Major Structures	3	4	4	5
Rotor Support System	Piggyback Bearing	Overhung Turbine/ Piggyback Bearing	Piggyback Bearing/ Hot Strut	Overhung Turbine/ Hot Strut
Overall Length, cm (in)	Base 182 (72)	30 (12)	7 (+3)	-15 (-6)
Maximum Diameter, cm (in)	Base 68 (27)	10 (+4)	Base	10 (+4)
Modularity (Number of Modules)	6	5	9	7
Deterioration Modes	Base	Least	Most	Same as STS646
Timeliness of Technology	Base	Greater Risk	Greatest Risk	Same as STS646
Reliability	Base	Base	Lowest	Lower
Component Performance and Matching	Base	Simpler	Simpler	Same as STS646
Starting Requirements	Base	55%	-45%	Same as STS646
Operating Constraints	Base	Fewer	Fewer	Same as STS646
Variable Geometry Complexity	Base	Least	Base	Same as STS646
Accel/Decel Time	Base	Lowest	Lower	Same as STS646
Prop-Fan Matching	Base	Improved Flexibility	Improved Flexibility	Improved Flexibility
Specific Fuel Consumption				
New	Base	+0.3%	+0.4%	+4.5%
After Approximately 3500 Cycles	Base	-0.3%	+0.4%	+3.9%
Engine Weight	Base	+7%	+13%	+13%
Engine Cost	Base	-7%	-1%	-4%
Engine Maintenance Cost	Base	-16%	-6%	-15%
Fuel Burn				
Percent Difference	Base	+0.6	+1.1	+5.6
After Approximately 3500 Cycles	Base	-0.1	+1.1	+4.9
Direct Operating Cost				
Percent Difference	Base	-1.5	-0.1	+0.8
After Approximately 3500 Cycles	Base	-1.7	-0.1	+0.6

Fuel burn and direct operating cost are the most important factors in the configuration ranking. Engine performance (thrust specific fuel consumption and pod drag) and weight are integrated to obtain the fuel burn comparison. Several parameters including performance, weight, acquisition cost, and maintenance cost contribute to the direct operating cost evaluation. Factors which were judged to be either comparable or of a second order of importance are listed in Table 4.2-XXX.

TABLE 4.2-XXX
ENGINE CONFIGURATION EVALUATION FACTORS
(Judged Comparable or of Second Order Influence)

Mechanical Design and Analysis Related Issues

- Turbine Cooling Requirements
- Future Engine Growth Paths
- Impact of Engine Design Choices on Propeller, Gearbox, Accessories, and Oil Cooling

Performance Related Issues

- Pod Drag (Specific Fuel Consumption)*
- Overall Pressure Ratio
- Turbine Rotor Inlet Temperature
- Power Turbine Work Extraction
- Propeller Drag in Connection with Failure Modes
- Effects of Anticipated Customer Bleeds and/or Horsepower Extraction
- Future Engine Growth Paths
- Impact of Engine Design Choices on Propeller, Gearbox, Accessories, and Oil Cooling

Environmental Related Issues (Noise and Pollution)

- Engine Noise Considerations
- Engine Emissions Considerations

* Except for Reversed Engine (STS646R) which is judged to have a higher pod drag

The practical scaling range for each of the four configurations was also evaluated. Results are summarized in Table 4.2-XXXI. The smallest practical size for the axial flow two and three-spool engines (STS648 and STS647) is 10,000 horsepower, due to rotor dynamics considerations and compressor blade size. The three-spool axial/centrifugal and reversed engine configurations (STS646 and STS646R) have a practical upper limit of 16,000 horsepower. This limit is imposed by centrifugal compressor performance, weight, low cycle fatigue life, and manufacturing considerations. These scaling ranges cover the engine sizes required for 90 - 150 passenger aircraft (refer to Figure 4.2-19).

TABLE 4.2-XXXI
PRACTICAL HORSEPOWER SCALING RANGE

Two-Spool Axial (STS648)	10,000 to 23,000 horsepower
Three-Spool Axial (STS647)	10,000 to 23,000 horsepower
Three-Spool Axial/Centrifugal (STS646)	8000 to 16,000 horsepower
Reversed Axial/Centrifugal (STS646R)	8000 to 16,000 horsepower

4.2.3.3 Configuration Update with Optimum Cycle

The engine configuration evaluation began with a screening study in which four candidate engine configurations were evaluated in a 10,000 horsepower base size with a design pressure ratio of 25:1. The two-spool axial compression engine and the three-spool axial/centrifugal compression engine were selected as the best candidates for a Prop-Fan propulsion system (Section 4.2.2.2). Because airframe manufacturers had not yet defined a final engine size for the reference 120-passenger aircraft, a final configuration evaluation was required to resolve critical issues relating to engine size and cycle.

With the two-spool axial compression configuration, mechanical arrangement, rotor dynamics, and critical speed become more significant problems as engine size is reduced: therefore, the axial compression engine was evaluated in the 10,000 horsepower size. With the three-spool axial/centrifugal configuration, centrifugal compressor design becomes more difficult as engine size is increased: therefore, the axial/centrifugal compression engine was evaluated in the 16,000 horsepower size. The cycle selected for each engine is listed in the top portion of Table 4.2-XXXII. Issues relating to mechanical design and aerothermodynamic analysis, performance, design assurance, and environmental constraints were updated for the resized engines. The key technical issues are also highlighted in Table 4.2-XXXII.

To select a final engine size, a propulsion system integration package, incorporating the updated engine size and cycle data, was submitted to the airframe manufacturers for review and comment. By mutual agreement with the participating airframe manufacturers and the NASA Program Manager, a final engine base size of 12,000 horsepower was selected. In addition to scaling the two engines to the final size, the overall cycle was modified to reflect the findings from the cycle optimization study (Section 4.2.1), as indicated in the bottom portion of Table 4.2-XXXII.

An interactive computer program was then used to generate an updated flowpath for each engine configuration. Cross sections were developed from these flowpaths and the characteristics of major components were defined. Materials, structural analysis and rotor dynamics issues were addressed, and engine performance was evaluated. Finally, the two engine configurations were compared on the basis of fuel burn and direct operating cost. Based on this evaluation and input from the participating airframe manufacturers, both the two-spool axial and the three-spool axial/centrifugal engine were selected for the Propulsion System Integration study (Task III).

TABLE 4.2-XXXII
SUMMARY OF UPDATED ENGINE SIZE AND AERODYNAMIC CYCLE

Configuration Update	"Non-Free" Power Turbine Engine[1] All-Axial Compression System	"Free" Power Turbine Engine[2] Axial/Centrifugal Compression System
o Engine Size, Maximum Horsepower	10,000	16,000
o Overall Compressor Pressure Ratio (Aerodynamic Cycle Design Point - 90% of Max Cruise Rating)	33.5	35
o Major Technical Issues	o Critical Speed (Rotor Dynamics) o Small Size Rear Stage Compressor Blades	o Performance, LCF Life, Structural Configuration and Manufacturing Limitations of Centrifugal Stage
Size, Cycle Selected for Propulsion System Integration Package		
o Engine Size, Max Horsepower	12,000	12,000
o Overall Compressor Pressure Ratio	34	34
o Technical Comparison	No one clear winner: Aircraft Companies Recommend Both Engine Concepts Be Included In Propulsion System Integration Package	

Notes:

1. A "non-free" power turbine engine is a configuration in which the low-pressure compressor is on the same spool as the power turbine which drives the Prop-Fan.

2. A "free" power turbine engine is a configuration in which the power turbine drives only the Prop-Fan.

4.2.3.3.1 Two-Spool Engine with All-Axial Compression System

The aerothermodynamic flowpath for the two-spool all-axial engine configuration, designated STS678 in the 12,000 horsepower base size, is shown in Figure 4.2-29. The flowpath features an eleven-stage high-pressure compressor driven by a two-stage high-pressure turbine. A four-stage power turbine drives a two-stage low compressor and the Prop-Fan through the gearbox (not shown on the flowpath). The flowpath, which is illustrated, provides a physical description of the path the air takes through the gas generator. Table 4.2-XXXIII provides a more detailed description of the engine, including a definition of the number of compressor and turbine stages, airfoil diameters, rotor speeds, number of airfoils, and airfoil geometry (for example aspect ratio and solidity). The aerodynamic loadings and materials selected reflect 1988 technology verification and 1992 engine certification.

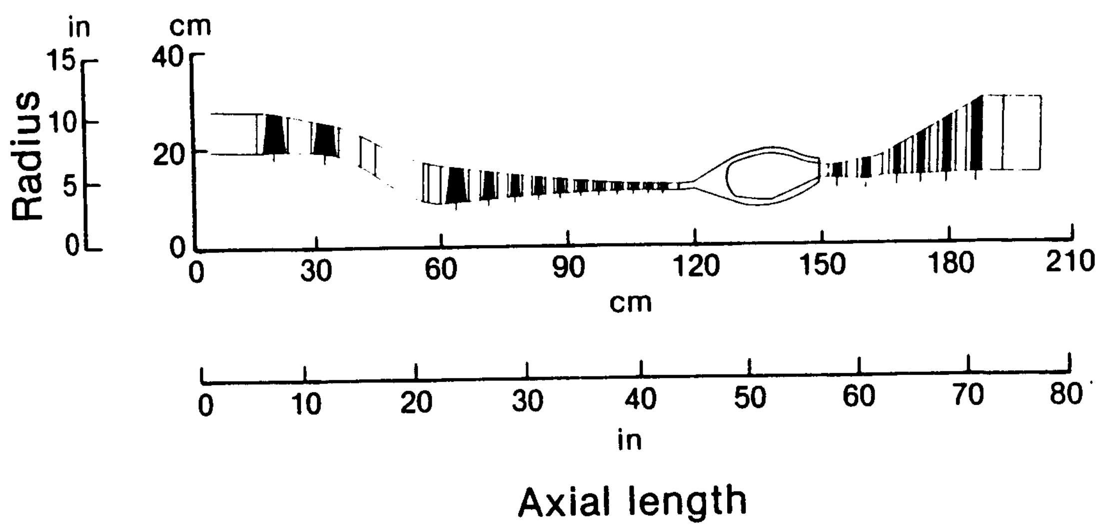

Figure 4.2-29 Aerothermodynamic Flowpath of the STS678 Engine - The updated flowpath for the 12,000 horsepower size two-spool axial compression engine was derived using 1988 technology for materials, cooling, and aerodynamics. (J27638-2)

The rotor speeds for both the high and low spools were set as high as possible without exceeding maximum turbine blade attachment stresses, resulting in the maximum turbine AN^2 limits noted in Table 4.2-XXXIII. The flowpath for the high spool is an extension of the engine technology studies conducted under the Energy Efficient Engine program. As indicated in the table, the high compressor pressure ratio is 17:1 using eleven stages.

An advanced single-stage aerating (MARK) burner concept is used in the combustor section.

TABLE 4.2-XXXIII

COMPONENT SUMMARY FOR THE TWO-SPOOL ALL-AXIAL COMPRESSION ENGINE
((STS678) 12,000 Horsepower Base Size)

Aerodynamic Design Cycle Parameters	
Altitude, m (ft)/Mach Number	10668 (35000)/0.75
Power Setting	90% Maximum Cruise
Overall Pressure Ratio	34
Max Combustor Exit Temperature (SLS, Hot Day), °C (°F)	1427 (2600)
Shaft Horsepower	12,000
Low Compressor	
Pressure Ratio	2.0
Number of Stages	2
Corrected Tip Speed, m/sec (ft/sec)	581 (1280)
Gap/Chord Ratio	0.60
Aspect Ratio	1.2
Number of Airfoils	162
Inlet Hub/Tip Ratio	0.69
Blade Tip Clearance, mils	11
Adiabatic Efficiency	87.2
High Compressor	
Pressure Ratio	17
Number of Stages	11
Corrected Tip Speed, m/sec (ft/sec)	401 (1315)
Gap/Chord Ratio	0.97
Aspect Ratio	1.5
Number of Airfoils	1059
Inlet Hub/Tip Ratio	0.53
Blade Tip Clearance, mils	11
Adiabatic Efficiency	86
Combustor	
Combustor Exit Temperature (CET), °C (°F)	1,426 (2600)
Combustor Inlet Temperature (CIT), °C (°F)	580 (1077)
Type of Combustor	Advanced Single Stage Aerating
Cooling Air Flux, kg/sec-m^2 (lb/sec-in^2)	15.874 (0.024)
Pressure Loss, % P_T in	
Overall	3.2
Liner	2.0
Burning Length, cm (in)	17/15 (6.7/5.9)
Number of Fuel Injectors	14
Pattern Factor, $\frac{T_{MAX} - CET}{CET - CIT}$	0.20
Space Heat Release Rate, watts/Nm (Btu/hr ft^3 atm)	4.29×10^2 (4.2×10^6)
Emissions	Potential to Meet Regulations
Lean Blowout Fuel to Air Ratio	0.004
High-Pressure Turbine	
Expansion Ratio	4.4
Number of Stages	2
Mean Velocity Ratio	0.63
Maximum AN^2 (X 10^{10}) cm^2RPM^2(in^2RPM^2)	32.3 (5.0)
Number of Airfoils	154
Blade Tip Clearance, mils	11
Adiabatic Efficiency	90.2
Power Turbine	
Expansion Ratio	8.4
Number of Stages	4
Mean Velocity Ratio	0.60
Number of Airfoils	634
Maximum AN^2 (X 10^{10}) cm^2RPM^2(in^2RPM^2)	38.7 (6.0)
Blade Tip Clearance, mils	11
Adiabatic Efficiency, %	94.1

The low-pressure compressor is on the same spool as the four-stage power turbine; rotor speed is limited by the maximum turbine blade attachment stress (AN^2) of the last turbine stage. This results in a two-stage low compressor configuration with a corrected tip speed of 390 m/sec (1280 ft/sec), illustrated in Figure 4.2-29. The two-stage configuration provides sufficient radial space (diameter) for the bearing compartment. The low spool (low-pressure compressor and power turbine) was designed with variable compressor vanes to permit the Prop-Fan to operate at constant speed at critical off-design conditions including takeoff, maximum climb, and cruise.

The four-stage power turbine configuration provides the velocity ratio required for a highly efficient close coupled turbine arrangement.

The cross section for the STS678 engine is presented in Figure 4.2-30; mechanical design features are summarized in Table 4.2-XXXIV.

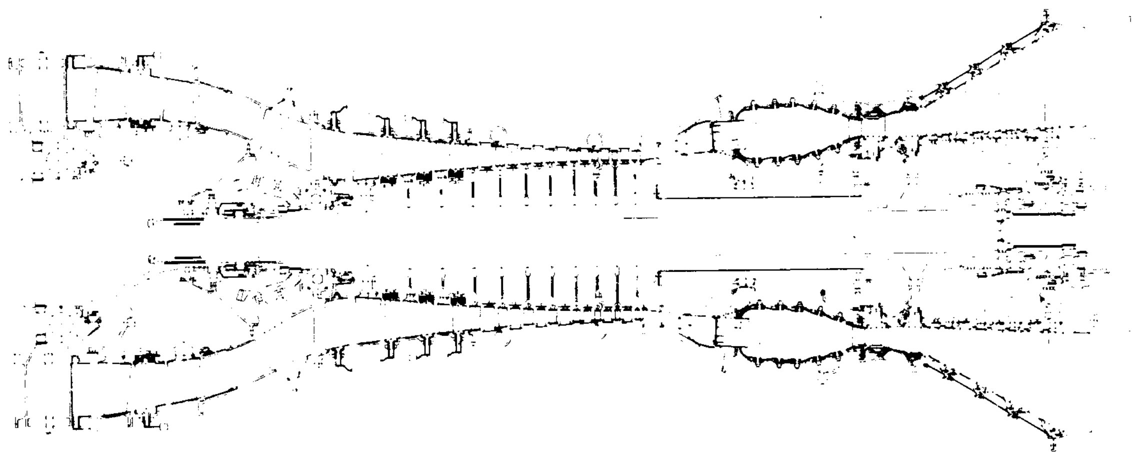

Figure 4.2-30 Mechanical Cross Section of the Updated Two-Spool Axial Compression Engine (STS678) - This 12,000 horsepower size engine features high rotor speeds for both spools and a low spool (low-pressure compressor and power turbine) designed to permit constant Prop-Fan speed at the critical off-design operating conditions. (J27638-4)

Three-Spool Engine with Axial/Centrifugal Compression

The aerothermodynamic flowpath for the three-spool axial/centrifugal engine configuration, designated STS679 in the 12,000 horsepower size, is shown in Figure 4.2-31.

The high spool is an axial/centrifugal compression system driven by a single-stage high-pressure turbine. The high-pressure compressor system features two axial compression stages followed by a single centrifugal compression stage. A pipe diffuser is used and a single-stage aerating burner is canted to mate with the centrifugal compressor.

TABLE 4.2-XXXIV
MECHANICAL DESIGN FEATURES OF THE TWO-SPOOL AXIAL
(Compression Engine (STS678) 12,000 Horsepower Base Size)

Inlet Case

- Cast aluminum, integral inner case, vane and outer case construction
- Provides bearing support for low compressor and power turbine shaft

Two-Stage Low Compressor

- Two-stage one piece rotor construction
- Variable stators for trailing edge of first stage vane and second stage vane
- Split outer case

Intermediate Case

- Houses Number 2 (rear of low compressor) and Number 3 (high compressor) thrust bearings
- Provides for engine accessory drive system

Eleven-Stage High Compressor

- Four variable vane stages
- Eleven stages separated into (a) front hub and disk, (b) rear one piece rotor

Diffuser/Combustor/High-Pressure Turbine Vanes

- Pipe diffuser to provide high-pressure turbine cooling air (TOBI)
- Pin fin combustor liner construction
- Cantilevered combustor support system

Two-Stage High-Pressure Turbine

- Full ring sideplates, wire seals for reduced leakage
- Simplified interstage seal arrangement
- Second stage cooling air supplied through second vane
- Segmented ceramic coated outer air seals

Four-Stage Power Turbine and Piggyback Bearing

- Four-stage unitized rotor construction
- Shrouded blades
- One piece welded construction
- Piggyback damped bearing support system

The intermediate spool consists of a four-stage low-pressure compressor driven by a single-stage intermediate turbine.

The three-stage power turbine, located on a third concentric shaft, provides shaft horsepower to the Prop-Fan through the gearbox (not shown on the flowpath).

The aerodynamic loadings and materials selected for the flowpath definition reflect 1988 technology verification and 1992 engine certification.

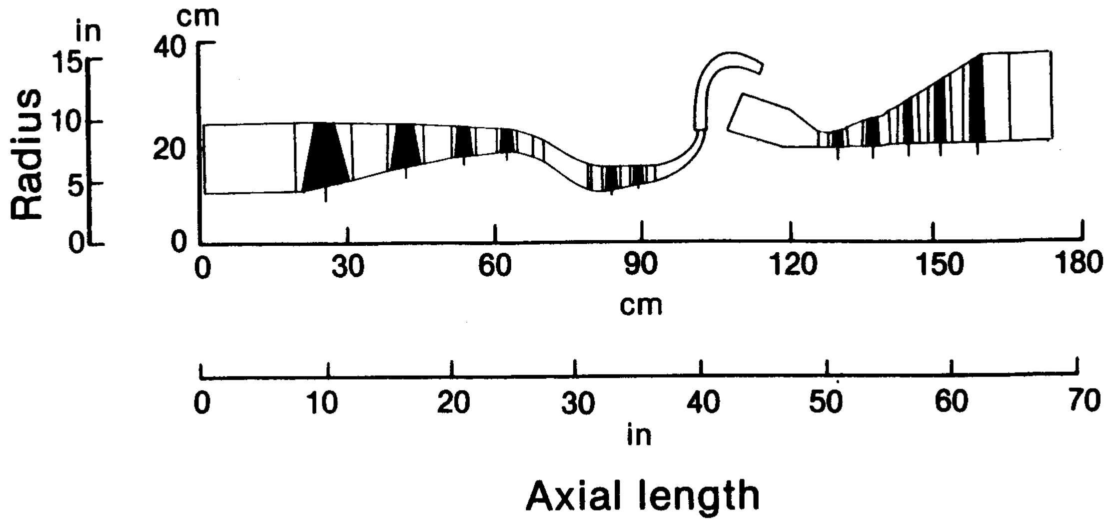

Figure 4.2-31 Aerothermodynamic Flowpath of the STS679 Engine - The updated flowpath for the 12,000 horsepower size three-spool axial/centrifugal compression engine was derived using 1988 technology for materials, cooling, and aerodynamics. (J27638-1)

The major components for the STS679 engine are described in Table 4.2-XXXV. The pressure ratio split between the high and low spool permits a close coupled high/intermediate power turbine configuration without exceeding turbine loading levels (velocity ratio), maximum turbine AN^2, acceptable compressor tip speeds, number of turbine stages, and maximum turbine rim speeds.

The selection of the high spool rotor speed was based on the maximum high-pressure turbine blade attachment stress (AN^2). The velocity ratio in the high-pressure turbine of the three-spool engine is slightly lower than the velocity ratio in the high-pressure turbine of the two-spool engine. This difference results from the 620 m/sec (2035 ft/sec) maximum rim speed in the centrifugal compressor, set by limitations in materials technology for the 1988 technology verification date. The axial/centrifugal pressure ratio split in the two-stage axial/single-stage centrifugal compressor was chosen to maximize efficiency.

TABLE 4.2-XXXV
COMPONENT SUMMARY FOR THE THREE-SPOOL AXIAL/CENTRIFUGAL ENGINE (STS679)

Aerodynamic Design Cycle Parameters		
Altitude, m (ft)/Mach Number		10668 (35000)/0.75
Power Setting		94% Maximum Cruise
Max Combustor Exit Temperature (SLS, Hot Day), °C (°F)		1427 (2600)
Overall Pressure Ratio		34
Shaft Horsepower, SLS - Hot Day		12,000
Low Compressor		
Pressure Ratio		5.4
Number of Stages		4
Corrected Tip Speed, m/sec (ft/sec)		658 (1450)
Gap/Chord Ratio		0.59
Aspect Ratio		1.2
Number of Airfoils		310
Inlet Hub/Tip Ratio		0.44
Blade Tip Clearance, mils		11
Adiabatic Efficiency		88.1
High Compressor	**Axial**	**Centrifugal**
Pressure Ratio	1.76	3.6
Number of Stages	2	1
Corrected Tip Speed, m/sec (ft/sec)	463 (1020)	--
Maximum Exit Wheel Speed, m/sec (ft/sec)	--	923 (2035)
Gap/Chord Ratio	1.0	--
Aspect Ratio	1.8	--
Number of Airfoils	139	--
Blade Tip Clearance, mils	11	Not Applicable
Adiabatic Efficiency	86.5	86.5
Combustor		
Combustor Exit Temperature (CET), °C (°F)		1,426 (2600)
Combustor Inlet Temperature (CIT), °C (°F)		580 (1077)
Type of Combustor		Canted Single Stage Aerating
Cooling Air Flux, kg/sec-m^2 (lb/sec-in^2)		16.874 (0.024)
Pressure Loss, % P_T in		
Overall		--
Liner		3.0
Burning Length, cm (in)		15 (5.9)
Number of Fuel Injectors		24
Pattern Factor, $\frac{T_{MAX} - CET}{CET - CIT}$		0.24
Space Heat Release Rate, Watts/Nm (Btu/hr ft^3 atm)		4.08×10^2 (4.0×10^6)
Emissions		Potential to Meet Regulations
Lean Blowout Fuel to Air Ratio		0.0095/0.0066 (Staged)
High-Pressure Turbine		
Expansion Ratio		3.3
Number of Stages		1
Mean Velocity Ratio		0.61
Maximum AN^2 (X 10^{10}) cm^2RPM^2(in^2RPM^2)	32.3	(5.0)
Number of Airfoils		75
Blade Tip Clearance, mils		11
Adiabatic Efficiency		89.7
Intermediate Turbine		
Expansion Ratio		2.0
Number of Stages		1
Mean Velocity Ratio		0.58
Maximum AN^2 (X 10^{10})		3.6
Number of Airfoils		89
Blade Tip Clearance, mils		11
Adiabatic Efficiency		89.7
Power Turbine		
Expansion Ratio		6.1
Number of Stages		3
Mean Velocity Ratio		0.60
Number of Airfoils		507
Maximum AN^2 (X 10^{10}) cm^2RPM^2(in^2RPM^2)	38.7	(6.0)
Blade Tip Clearance, mils		11
Adiabatic Efficiency		94.1

The intermediate spool rotor speed was limited by the low-pressure compressor corrected tip speed of 438 m/sec (1440 ft/sec) which was considered a reasonable trade between efficiency, weight, and cost. This tip speed, coupled with the requirement to provide sufficient radial space for the bearing compartments, led to selection of an intermediate turbine rotor speed below the maximum AN^2 limit.

The three-stage power turbine configuration is used to achieve the velocity ratio required for high efficiency in a close coupled mechanical arrangement in which the speed is set by the maximum turbine blade attachment stress in the last stage.

The cross section for the STS679 engine is presented in Figure 4.2-32; mechanical design features are summarized in Table 4.2-XXXVI.

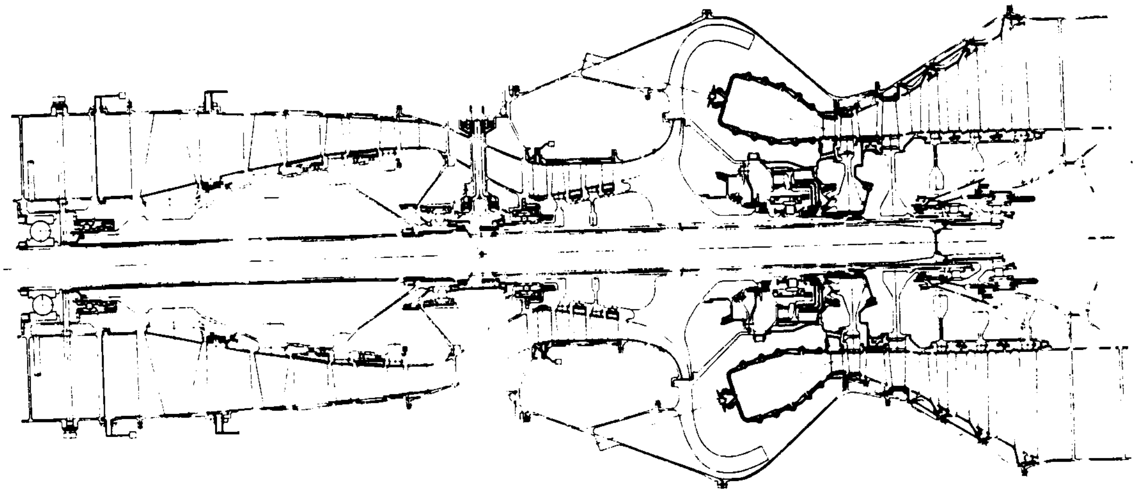

Figure 4.2-32 Mechanical Cross Section of the Updated Three-Spool Axial/Centrifugal Compression Engine (STS679) - This 12,000 horsepower size engine features a close coupled high/intermediate/power turbine configuration designed to maximize efficiency and minimize engine weight and cost. (J27638-5)

Summary of Mechanical Design and Analysis Issues

Updated mechanical design and analysis issues are summarized in Table 4.2-XXXVII. The parameters which were modified include: bearing DN levels, overall length, diameter, and number of modules. However, these changes did not have a major impact on the overall comparison of the two configurations. Factors judged comparable or of a second order of importance include turbine cooling requirements, future engine growth paths, and the impact of design choices on the Prop-Fan, gearbox, accessories, and oil cooling.

TABLE 4.2-XXXVI
MECHANICAL DESIGN FEATURES OF THE THREE-SPOOL AXIAL/CENTRIFUGAL ENGINE
((STS679) 12,000 Horsepower Base Size)

Inlet Case

- Cast aluminum, integral inner case, vane and outer case construction
- Provides bearing support for low compressor and power turbine shaft

Four Stage Low Compressor

- Four-stage one piece welded rotor
- Two variable stators
- Split outer case

Intermediate Case

- Contains Number 3 and Number 4 damped bearings
- Provides for engine accessory drive system

Three-Stage High Compressor

- First vane is variable
- Two axial stages integral with centrifugal stage
- Split internal case, one piece outer case is cantilevered construction

Diffuser/Combustor/Bearing Support/High Pressure Turbine Vanes

- Pipe diffuser
- Pin fin combustor louver construction
- Cantilevered combustor support system
- Damped bearing arrangement
- Pressurized labyrinth seals

Single-Stage High-Pressure Turbine

- Full ring sideplates, wire seals for reduced leakage
- Simplified interstage seal arrangement
- Segmented ceramic coated outer air seals

Single Stage Intermediate Turbine

- Full ring sideplates, wire seals for reduced leakage
- Simplified interstage seal arrangement
- Rotor cooling air supplied externally through turbine nozzle vane
- Segmented ceramic cooled outer air seals

Three-Stage Power Turbine and Piggyback Bearing

- Three stage unitized rotor construction
- Shrouded blades
- One piece welded construction
- Piggyback damped bearing support system

TABLE 4.2-XXXVII
UPDATE OF MECHANICAL DESIGN AND ANALYSIS ISSUES

	Two-Spool All-Axial Compression System		Three-Spool Axial/Centrifugal Compression System	
	Initial Assessment STS648	Update STS678	Initial Assessment STS646	Update STS679
Compressor Pressure Ratio, Low/High	2/12.5	2/17	7.1/3.6	5.3/6.3
Engine Size, Horsepower	10,000	12,000	10,000	12,000
Compressor Stages	3 + 12	2 + 11	5 + 1	4 + 3
Turbine Stages	2 + 4	2 + 4	1 + 1 + 3	1 + 1 + 3
Number of Bearings	5	5	7	7
Max Bearing DN (Millions)	2.3	2.5	2.4	2.8
Bearing Supports and Major Structures	3	3	4	4
Rotor Support System	PB	PB	OH/PB	OH/PB
Overall Length, cm (in)	Base	Base	-30 (-12)	-12 (-5)
Maximum Diameter, cm (in)	Base	Base	+10 (+4)	Up to 5% lower
Modularity (Number of Modules)	6	6	5	7
Deterioration Modes	Base	Base	Least	Lower
Timeliness of Technology	Base	Base	Greater Risk	Greater Risk

OH = Overhung Turbine   PB = Piggyback Bearings

4.2.3.3.2 Performance Related Issues

Based on comments from the airframe manufacturers, the turboprop engine was rerated to more closely reflect the aircraft thrust requirements at the takeoff, climb, and cruise conditions. The original and revised ratings are compared in Table 4.2-XXXVIII. The revised ratings were achieved by a combination of reduced throttle operation at takeoff and a hot section rematch. These changes reduced engine core size and takeoff shaft horsepower but maintained takeoff combustor exit temperature.

TABLE 4.2-XXXVIII
TURBOPROP ENGINES RERATED TO IMPROVE
COMPATIBILITY WITH AIRCRAFT REQUIREMENTS

	Original Ratings	Revised Ratings
Aerodynamic Design Point		
Design Inlet Corrected Flow, kg/sec (lb/sec)	28.4 (62.5)	27.3 (60.1)
Design Overall Pressure Ratio	34	34
Core Size, Compressor Exit Corrected Airflow, kg/sec (lb/sec)	1.4 (3.18)	1.4 (3.06)
Maximum Cruise Rating		
Maximum Cruise Combustor Exit Temp, °C (°F)	1221 (2230)	1221 (2330)
Maximum Cruise TSFC	Base	+0.3%
Maximum Climb Thrust, N (lb)	17,881 (4020)	17,881 (4020)
Maximum Climb Overall Pressure Ratio	38.3	38.3
Combustor Exit Temperature, STD +10°C (+18°F)	1332 (2430)	1388 (2530)
Prop-Fan Loading, shp/D^2	34.2	34.2
Prop-Fan Diameter, m (ft)	4.05 (13.3)	4.05 (13.3)
Takeoff Rating		
Takeoff Thrust @ M=0.22, STD +14°C (+25°F) Day, N (lb)	74,507 (16750)	69,836 (15700)
Maximum Shaft Horsepower	13300	12000
Combustor Exit Temperature, °C (°F)	1388 (2530)	1388 (2530)
Prop-Fan Maximum Loading, shp/D^2	74.7	67.4

The updated performance related issues are summarized in Table 4.2-XXXIX. While the performance comparison differs from the original assessment, the changes are not considered significant in the overall evaluation of the two engines. The following factors were judged to be comparable or of a second order of importance in the performance evaluation: pod drag, SFC, overall pressure ratio, turbine rotor inlet temperature, power turbine work extraction, propeller drag in failure mode, effects of anticipated customer bleeds and/or horsepower extraction, future engine growth paths, impact of engine design choices on the Prop-Fan, gearbox, and oil cooling system.

TABLE 4.2-XXXIX
UPDATE OF PERFORMANCE RELATED ISSUES

	Two-Spool All-Axial Compression System		Three-Spool Axial/Centrifugal Compression System	
	Initial Assessment STS648	Update STS678	Initial Assessment STS646	Update STS679
Compressor Pressure Ratio, Low Compressor/ High Compressor	2/12.5	2/17	7.1/3.6	5.4/6.3
Engine Size, Horsepower	10,000	12,000	10,000	12,000
Component Performance and Matching	Base	Base	Simpler	Simpler
Starting Requirements	Base	Base	-55%	-45%
Operating Constraints	Base	Base	Fewer	Fewer
Variable Geometry Complexity	Base	Base	Least	Lower
Accel/Decel Time	Base	Base	Lowest	Lower
Prop-Fan Matching	Base	Base	Improved Flexibility	Improved Flexibility
New SFC (After Approx 3500 Cycles	Base (Base)	Base (Base)	+0.3%(-0.3%)	+1% (+1/2%)

4.2.3.3.3 Design Assurance Related Issues

Updated design assurance related issues are summarized in Table 4.2-XL. While there are some differences from the initial assessment, they are not considered significant.

4.2.3.3.4 Environmental Issues

Both updated engine configurations have the potential to satisfy the noise and emissions goals specified for the APET Program.

TABLE 4.2-XL
UPDATE OF DESIGN ASSURANCE RELATED ISSUES

	Two-Spool All Axial Compression System		Three-Spool Axial/Centrifugal Compression System	
	Initial Assessment STS648	Update STS678	Initial Assessment STS646	Update STS679
Compressor Pressure Ratio (Low Compressor/High Compressor)	2/12.5	2/17	7.1/3.6	5.4/6.3
Engine Size, Horsepower	10,000	12,000	10,000	12,000
Reliability	Base	Base	Base	Base
Engine Weight	Base	Base	+7%	+9%
Engine Cost	Base	Base	-7%	-8%
Engine Maintenance Cost	Base	Base	-16%	-12%

4.2.3.3.5 Summary of Results

Table 4.2-XLI summarizes the key findings from the evaluation of the updated mechanical design and analysis issues, performance related issues, and design assurance related issues for the two 12,000 horsepower size engines. The fuel burn and direct operating costs of the two engines were then compared, using aircraft trade factors for specific fuel consumption, engine weight, price, and maintenance cost. The two-spool axial compression engine demonstrated a fuel burn advantage, while the three-spool axial/centrifugal engine exhibited lower direct operating costs. Based on these results, and input from the airframe manufacturers, both engine configurations were selected for the Propulsion System Integration Package.

4.2.4 Technical Considerations Requiring Additional Study Effort Beyond the Scope of the Current Contract

The engine configuration evaluation led to the identification of several technical considerations for which additional study effort, beyond the scope of the current contract, is recommended. These issues are summarized in Table 4.2-XLII and discussed briefly below.

TABLE 4.2-XLI
CONFIGURATION EVALUATION SUMMARY

Engine Configuration	STS678 Axial Two-Spool	STS679 Axial/Centrifugal Three-Spool
Compressor Pressure Ratio	2 X 17	5.4 X 6.3
Engine Size, Horsepower	12,000	12,000
Compressor Stages	2 + 11	4 + (2 + 1)
Turbine Stages	2 + 4	1 + 1 + 3
Number of Bearings	5	7
Maximum Bearing DN (millions)	2.5	2.8
Bearing Supports and Major Structures	3	4
Rotor Support System	PB	OH/PB
Overall Length, cm (in)	Base	-12 (-5)
Maximum Diameter, cm (in)	Base	Base
Modularity (Number of Modules)	6	7
Deterioration Modes	Base	Lower
Timeliness of Technology	Base	Greater Risk
Reliability	Base	Up To 5% Lower
Component Performance and Matching	Base	Simpler
Starting Requirements	Base	-45%
Operating Constraints	Base	Fewer
Variable Geometry Complexity	Base	Lower
Accel/Decel Time	Base	Lower
Prop-Fan Matching	Base	More Flexibility
o New SFC (After About 3500 Cycles)	Base (Base)	+1% (+ 1/2%)
o Engine Weight	Base	+9%
o Engine Cost	Base	-8%
o Engine Maintenance Cost	Base	-12%
Fuel Burn (After 3500 Cycles)	Base (Base)	+1.5% (+0.9%)
Direct Operating Cost (After 3500 Cycles)	Base (Base)	-0.9% (-1.1%)

OH = Overhung Turbine PB = Piggyback Bearings

Flowpath dirt removal and bird ingestion requirements must be defined for both the two-spool axial compression engine and the three-spool axial/centrifugal compression engine; innovative systems must be developed in response to these requirements.

Finally, there are operational considerations for the "free" turbine (three-spool) and "non-free" turbine (two-spool) engines which must be addressed in joint studies with the airframe manufacturers.

TABLE 4.2-XLII
ENGINE CONFIGURATION TECHNICAL CONSIDERATIONS
REQUIRING ADDITIONAL STUDY

- Define Specific Requirements and Solutions for Flowpath Dirt Removal
- Define Specific Bird Ingestion Requirements and Potential Solutions
- Estimate and Compare "Free" (Three-Spool) vs "Non-Free" (Two-Spool) Power Turbine Operating Characteristics to Determine Impact on Transient Aircraft Operating Characteristics

SECTION 4.3 -- DISCUSSION OF RESULTS
Task III -- Propulsion System Integration

4.3 TASK III - PROPULSION SYSTEM INTEGRATION

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List of Tables for Section 4.3 (Cont'd)

4.3 TASK III - PROPULSION SYSTEM INTEGRATION

4.3.1 Introduction

The objective of Task III was to select the best propulsion system for a Prop-Fan powered aircraft and to prepare a Propulsion System Integration Package which would be used to compare the Prop-Fan and reference turbofan propulsion systems in the Engine/Aircraft Evaluation (Task IV).

At the conclusion of Task II, the NASA Program Manager and Pratt & Whitney mutually selected two engine configurations for the propulsion system integration studies: (1) a "non-free" power turbine (two-spool) engine with axial compression and (2) a "free" turbine (three-spool) engine with axial/centrifugal compression.

Initial Propulsion System Integration Packages were assembled and submitted to the four airframe manufacturers participating in the APET Study (Boeing, Douglas, Lockheed-California and Lockheed-Georgia) for their critique and comments. The integration packages included an over-the-wing installation for the two-spool and three-spool engine configurations, in-line and offset reduction gear options, gearbox oil cooler information, inlet configuration options and comparisons, and a propulsion system integration summary. Pratt & Whitney received written comments from Lockheed-Georgia, Lockheed-California, and Boeing. The comments made by the participating airframe manufacturers were incorporated in the Final Propulsion System Integration Package discussed in this section. Some of the major features of the final integration package include: confirmation of an over-the-wing installation, a turboprop base engine size of 12,000 horsepower, and confirmation that the propulsion system integration package should continue to include options for "free" and "non-free" power turbine engine configurations, and in-line and offset reduction gear concepts. During Task III, follow-on work was identified which will require coordinated efforts between engine and airframe manufacturers.

This section covers the following topics in some detail: (1) propulsion system component definition, including engine configurations, gearbox and pitch control candidates, aircraft accessory locations for power extraction, inlet configurations, oil cooler arrangements, propulsion system control, and propeller (Prop-Fan) considerations; (2) the major features of the integrated propulsion system, including a conceptual nacelle, engine mounting options, acoustic treatment requirements, modular maintenance concepts, and propulsion system reliability; (3) a summary of the Final Propulsion System Integration Package. The integration package includes: a conceptual drawing of the two propulsion systems, a base size turboprop/reference turbofan engine comparison, turboprop engine system data package and computer deck (including User Manual), and a reference turbofan engine propulsion system data package and computer deck (including user manual). The user manuals for both the turboprop and reference turbofan engines include weight, dimensional, and performance scaling curves as a function of engine size. Acquisition and maintenance cost data, as well as appropriate scaling curves, will be supplied in a separate document.

4.3.2 Propulsion System Components

The propulsion system components include: the turboprop engine, reduction gearbox/pitch control, aircraft accessory locations for power extraction, engine inlet configuration with options, oil cooler with options, an engine/Prop-Fan control system, and the Prop-Fan. The process of evaluating these components and defining the final propulsion system configuration is described below.

4.3.2.1 Turboprop Engine

As a result of the cycle optimization study and engine configuration evaluation (Task II), both the two-spool axial compression engine (designated STS678 in the 12,000 horsepower base size) and the three-spool axial/centrifugal engine (designated STS679 in the 12,000 horsepower base size) were selected for the propulsion system integration study. The "non-free" power turbine engine (STS678) is illustrated in Figure 4.3-1 and the "free" power turbine engine (STS679) is illustrated in Figure 4.3-2. These two engine configurations were judged approximately equal on the basis of fuel burn and direct operating cost. The details of the selection process were presented in Section 4.2.

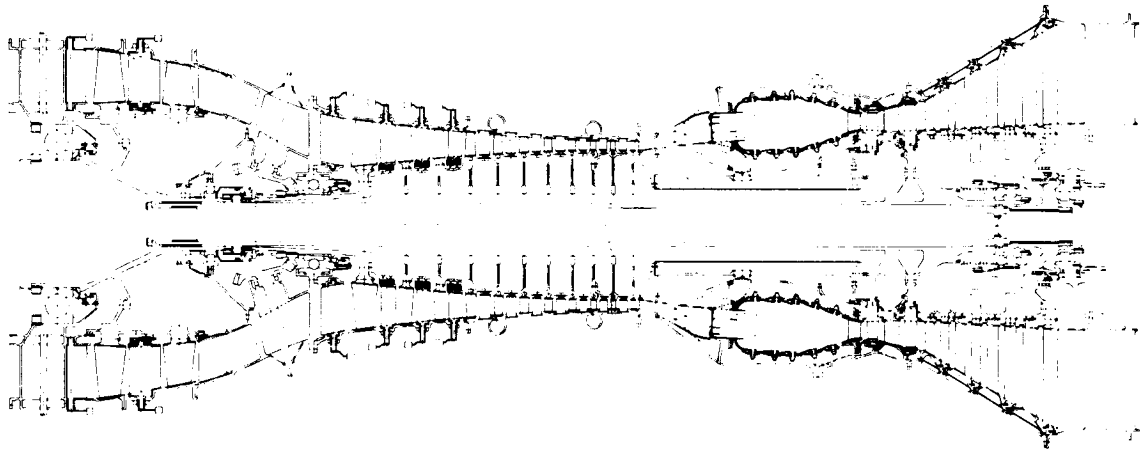


Figure 4.3-1 Cross Section of the STS678 Engine - This "non-free" power turbine (two-spool) axial compression engine was selected for evaluation in an integrated Prop-Fan propulsion system. (J27638-4)

4.3.2.2 Gearbox and Pitch Control

Pratt & Whitney conducted extensive studies of reduction gear systems in 1981. Information obtained in these studies is provided as background for the work conducted during the APET contract effort.

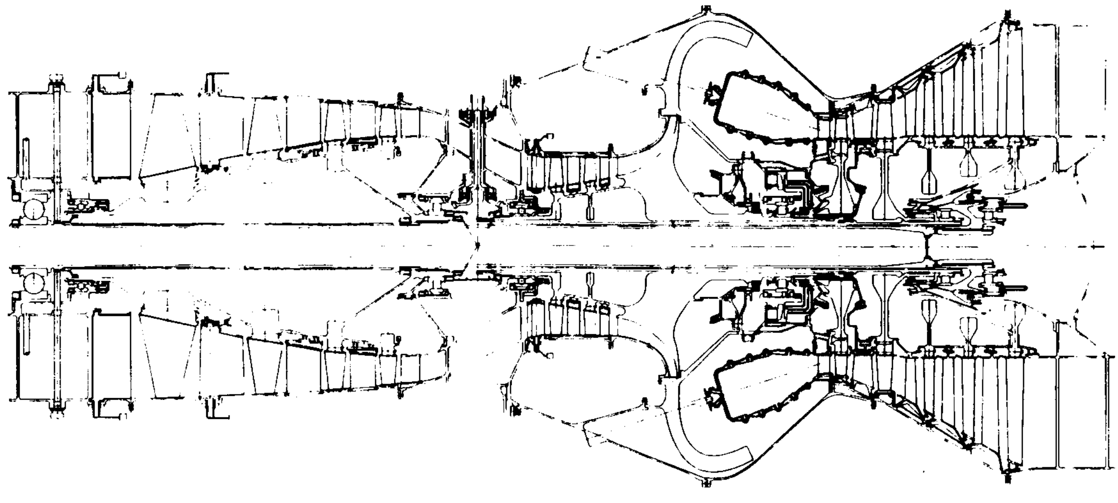


Figure 4.3-2 Cross Section of the STS679 Engine - This "free" power turbine (three-spool) axial/centrifugal compression engine was selected for evaluation in an integrated Prop-Fan propulsion system. (J27638-5)

Based on these studies, the in-line split path gearbox configuration and the offset compound idler gearbox configuration were selected for evaluation. Pitch control considerations were also covered. The airframe manufacturers indicated that both systems should be included in future engine/aircraft integration studies. Therefore, both the in-line split path configuration and the offset compound idler configuration are included in the Propulsion System Integration Package.

Background Information

The results of previous gearbox studies were documented in Technical Paper AIAA-82-1124, "Selecting the Best Reduction Gear Concept for a Prop-Fan Propulsion System," delivered at the June 1982 AIAA/SAE/ASME 18th Joint Propulsion System Conference. The major findings from the paper are summarized in Table 4.3-I. Nine in-line and offset gearbox concepts were screened in the studies. The in-line concepts were: split path planetary, layshaft, compound planetary, star/star, and planetary/planetary systems. The offset reduction gear concepts included: spur/star, spur/planetary, compound idler, and spur/spur systems. The systems were evaluated on the basis of efficiency, reliability, weight, acquisition cost, maintenance cost, Prop-Fan pitch control accessibility, risk assessment, capability of opposite rotation, ease of scaling, and noise generation.

TABLE 4.3-I
REDUCTION GEAR CONFIGURATION SYSTEM EVALUATION

15,000 HORSEPOWER SIZE

1981 TECHNOLOGY

REDUCTION GEAR CONFIGURATION	MAX. HORSEPOWER LOSS	CRUISE EFFICIENCY	WEIGHT, LB.	MFG COST, '80 $	MAINT COST,'80 $/EFH	RELATIVE SYSTEM DOC + INT
COMPOUND IDLER (OFFSET)	166	0.982	BASE	BASE	BASE	BASE
SPUR/PLANETARY (OFFSET)	172	0.981	-9%	+50%	+105%	+0.32%
SPLIT PATH (IN-LINE)	182	0.980	-30%	+50%	+100%	+0.09%
STAR/PLANETARY (IN-LINE)	213	0.977	-8%	+50%	+105%	+0.62%
SPUR/STAR (OFFSET)	213	0.977	+4%	+50%	+50%	+0.72%
STAR/STAR (IN-LINE)	224	0.976	-3%	+60%	+110%	+0.85%

As shown in the table, the offset compound idler and the in-line split path planetary gearbox concepts were found to be very close on the basis of direct operating cost. Therefore, both concepts were selected for further evaluation in the APET study using technology assumed to be available in the 1988 time period.

To ensure a comprehensive evaluation of the reduction gear configurations, a team of experts was assembled from United Technologies Corporation to take advantage of the Corporation's extensive experience with gearboxes for helicopters, small turboprop engines, and military applications. The makeup of the team is described in Table 4.3-II; key results are also highlighted.

TABLE 4.3-II
UNITED TECHNOLOGIES CORPORATION REDUCTION GEAR EVALUATION TEAM

<u>Design</u>

- o Pratt & Whitney - Commercial Engineering
- o Sikorsky Aircraft

<u>Design Review Team</u>

- o Hamilton Standard
- o Pratt & Whitney of Canada
- o Pratt % Whitney - Government Products Division

o Considered	5 In-line gear candidates 4 Offset gear candidates
o Compared	Weight, efficiency, durability, cost
o Selected	Split path in-line and compound idler offset configurations for further studies

The gearbox design objectives are summarized in Table 4.3-III. The objectives reflect the major concern of airline operators of turboprop engines; improving the 2000 to 3000 hour mean time between unscheduled removals of the gearbox module for cause characteristic of current reduction gears. The mean time between removals (MTBR) is indicative of the reliability of the gearbox and leads to the assessment of maintenance cost which will be discussed later in this section.

TABLE 4.3-III
REDUCTION GEAR DESIGN APPROACH

Overall Design Goals

- o Reliability (MTBR) - Greater than 15,000 hours
- o Cruise efficiency - 99.0%

Mounting Philosophy - Transfer Prop-Fan Loads Directly to Aircraft

- o Stiff reduction gear shafts to minimize deflections
- o Output gear located on output shaft where shaft slope = 0 degree
- o Stiff case to minimize deformations
- o Bearings and gears sized for long life

Modular Construction for High Aircraft Dispatch Reliability and Low Maintenance Cost

- o Externally mounted aircraft accessories
- o Accessible oil pump and condition monitoring systems
- o Propeller brake pad
- o "On-the-wing" main shaft seal replacement
- o Accessible propeller pitch control components

To assist in meeting the reliability goals, the gears for an advanced turboprop reduction gear system will be designed with allowable stresses which are 30%-40% lower than helicopter design criteria. This standard reflects the longer life cycle requirement for a turboprop gear system. The stress levels for the turboprop gears are compared to stress levels for helicopter gears in Figure 4.3-3.

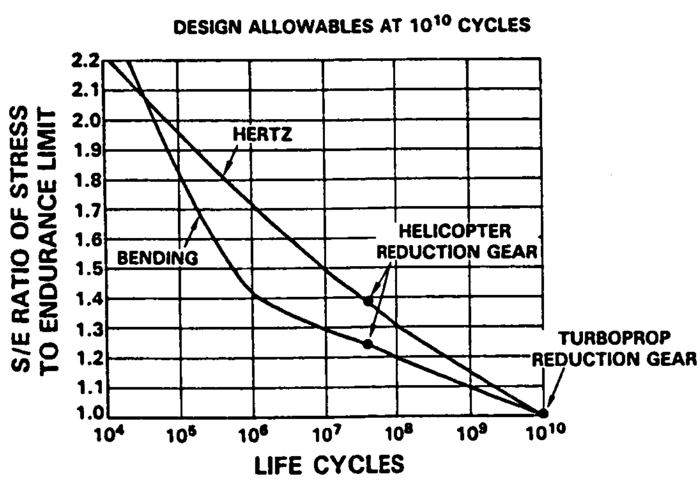


Figure 4.3-3 Spur-Helical Gear S-N Curve - The allowable gear stresses for a turboprop reduction gear system are significantly lower than helicopter design criteria. (J26767-6)

The offset compound idler and in-line split path reduction gear concepts were evaluated using technology assumed to be available by 1988. Current and advanced gear, bearing, housing, and lubricant technology levels are compared in Table 4.3-IV. To achieve the longer life design objective for advanced turboprop engine gearboxes, weight has been strategically added to the gearbox designs.

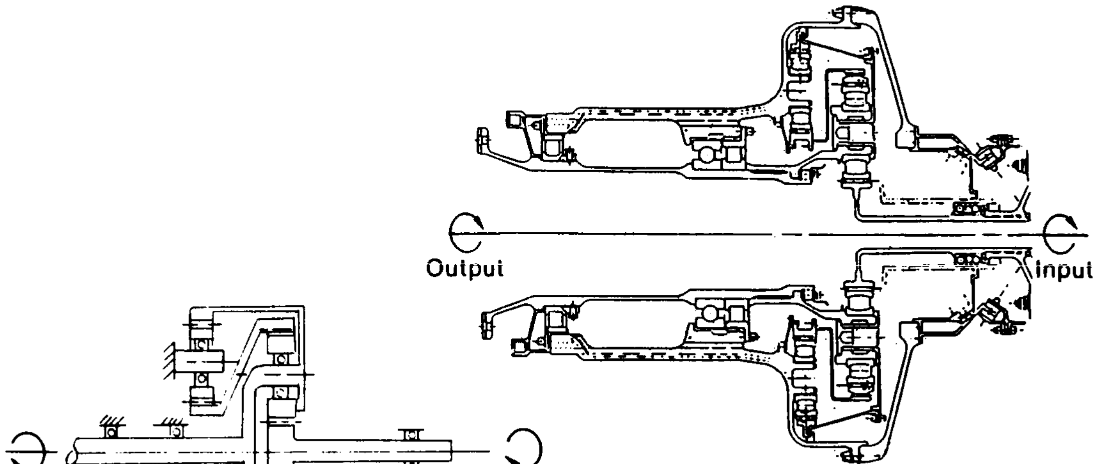
TABLE 4.3-IV
TECHNOLOGY FOR REDUCTION GEAR CONCEPT EVALUATION

	Current Technology	Advanced Technology Assumed Available by 1988
o Gears - Materials	AMS 6265	Vasco X-2M or Cartech EX-53
Bending Fatigue Limit		
Unidirectional, MPa	345(1)	414(1)
(psi)	(50,000)	(60,000)
Reversed Bending, MPa	283(1)	338(1)
(psi)	(41,000)	(49,000)
Hertz Stress Limit, MPa	869(1)	1041(1)
(psi)	(126,000)	(151,000)
Pitch Line Velocity Limit, m/min	9144	10,668
(ft/min)	(30,000)	(35,000)
o Bearings - Materials	CVM M50	Vimvar M50
System Design Life Requirement (L10) hr.	18,000	18,000
Material/Lubrication Life Factor	6-12	20 to 30
o Housings - Materials	Aluminum, Magnesium	Advanced Aluminum, Magnesium and/or Stainless Steel
o Lubricant - Fluids	Mil 23699 Type II	Synthesized Hydrocarbon Fluid (SHF)
Oil Inlet Temp., °C (°F)	82 (180)	121 (250)
Allowable Temperature Rise, °C (°F)	4.4-10 (40-50)	26.7-37.8 (80-100)
Load Carrying Ability, MPa (psi)	13.8-24.1 (2000-3500)	27.6-31.0 (4000-4500)
Flash Temperature Index, °C (°F)	136 (276)	177 (350)

(1)Typical gear allowable stress - 3 sigma with a coefficient of variation = 0.1, 10^{10} cycles

In-line Split Path Reduction Gear Configuration

The in-line split path reduction gear is a compact, lightweight system incorporating twenty gears and twenty-one bearings. A conventional rotation configuration is shown in Figure 4.3-4. A schematic of the gearbox is shown in the lower left corner of the figure. The overall gear (speed) ratio is 9.6. The speed reduction for the first and second stages is 5.2 and 1.85, respectively. Forty-four percent of the total power is transmitted through the first stage while the remaining 56% is transmitted through the second stage. Since the power transmission is split between both stages, smaller gears and bearings can be used. The second stage incorporates a star gear system which is located forward of the planetary first stage to improve access to the Prop-Fan pitch control modules. Although it is not shown in the figure, one accessory pad is provided for aircraft use and a Prop-Fan brake pad is included.



- 1 input gear, 2 bearings
- 1st stage
  - 6 pinion gears, 6 bearings
  - 1 ring gear
- 2nd stage
  - Input sun gear
  - 10 pinion gears, 10 bearings
  - 1 ring gear
  - Output shaft, 3 bearings

Figure 4.3-4 In-Line Split Path Reduction Gear Concept - This compact, lightweight system includes 20 gears and 21 bearings. (J27638-86)

In order to minimize maintenance costs, the gearbox has been designed as a series of subassemblies which can be removed as modules for inspection, repair, or replacement. Key features of this modular design are described in Table 4.3-V.

TABLE 4.3-V
GEARBOX SUBASSEMBLIES MADE REMOVABLE FOR
INSPECTION, REPAIR, OR REPLACEMENT

External Accessories
- Prop-Fan Pitch Control (all modules for offset gearbox concept; two of these modules for in-line gearbox concepts)
- Power Takeoff Shaft to Airframe Accessory Drive
- Prop-Fan Brake Assembly

Interface Components
- Input Shaft Coupling, Seal and Mating Ring
- Prop-Fan Shaft Seal and Mating Ring
- Mount Pads and Bushings

Lubrication System Components
- Oil Supply/Scavenge Pump Module(s)
- Oil Filter(s)
- Breather Filter
- Oil Jet Screens
- Oil Jets (two-thirds of total)

Condition Monitoring Components (Usually Supplied by Aircraft Company)

- Chip Detector Module(s)
- Vibration/Noise Monitoring Devices

Table 4.3-VI summarizes the individual bearing lives required to meet the total bearing system design objective of a B10 life of 18,000 hours. Spherical roller bearings were selected for the planet and star pinions to provide good bearing and gear alignment. The multiple load path design of the split path planetary gearbox concept results in a compact, lightweight system with twenty-one bearings. The bearing sizes selected to meet the design objectives will determine pinion gear diameter and face width.

TABLE 4.3-VI
BEARING LIFE SUMMARY IN-LINE, SPLIT PATH PLANETARY GEARBOX
(Conventional Rotation)

Bearing			Life, B10*
Location	Type	Quantity	Individual Bearing Hours
Input Shaft	Ball	2	10^6
Planet Pinion	Spherical Roller	6	135,000
Star Pinion	Spherical Roller	10	148,000
Output Shaft	Roller, Rear	1	105,000
Output Shaft	Roller, Front	1	100,000
Output Shaft	Ball	1	146,000
		21	
Bearing System Life			18,900

*Equivalent life that 90% of all bearing sets will meet or exceed

Gear stresses for the in-line split path planetary gearbox are summarized in Table 4.3-VII. The bearing sizes selected to meet the design objectives in Table 4.3-VI will determine pinion gear diameter and face width. The resulting pinion gear dimensions led to gear tooth Hertz contact stresses which are well below the 1041 MPa (151,000 psi) design allowable level. This will substantially reduce the probability of gear tooth pitting and scoring failures. To obtain maximum gear efficiency, the gear tooth pitch was selected such that the gear tooth bending stresses are at the design allowable level. The stresses setting the design constraints are associated with the bending limit for reverse loading in the pinion gear teeth.

TABLE 4.3-VII
GEAR STRESS SUMMARY IN-LINE, SPLIT PATH PLANETARY GEARBOX
(Conventional Rotation)

Gear		Stress				
Stage	Mesh	Type	Value MPa	Value (psi)	Allowable MPa	Allowable (psi)
Planet	Sun-Pinion	Hertz	861	(124,800)	1041	(151,000)
		Bending	332	(48,200)	338	(49,000)*
	Pinion-Ring	Hertz	544	(79,000)	1041	(151,000)
		Bending	299	(43,300)	338	(49,000)*
Star	Sun-Pinion	Hertz	886	(128,500)	1041	(151,000)
		Bending	278	(40,300)	338	(49,000)*
	Pinion-Ring	Hertz	712	(103,300)	1041	(151,000)
		Bending	271	(39,300)	338	(49,000)*

*Bending Limit Set by Reverse Loading of Pinion Gear Tooth

To provide some insight into the integration of the pitch control with the in-line gearbox, an enlarged view of this portion of the gearbox is shown in Figure 4.3-5. The differential gear assembly and hydraulic transfer bearing module of the pitch control are located within the gearbox. Therefore, the gearbox will have to be removed from the aircraft for pitch control maintenance actions in these areas.

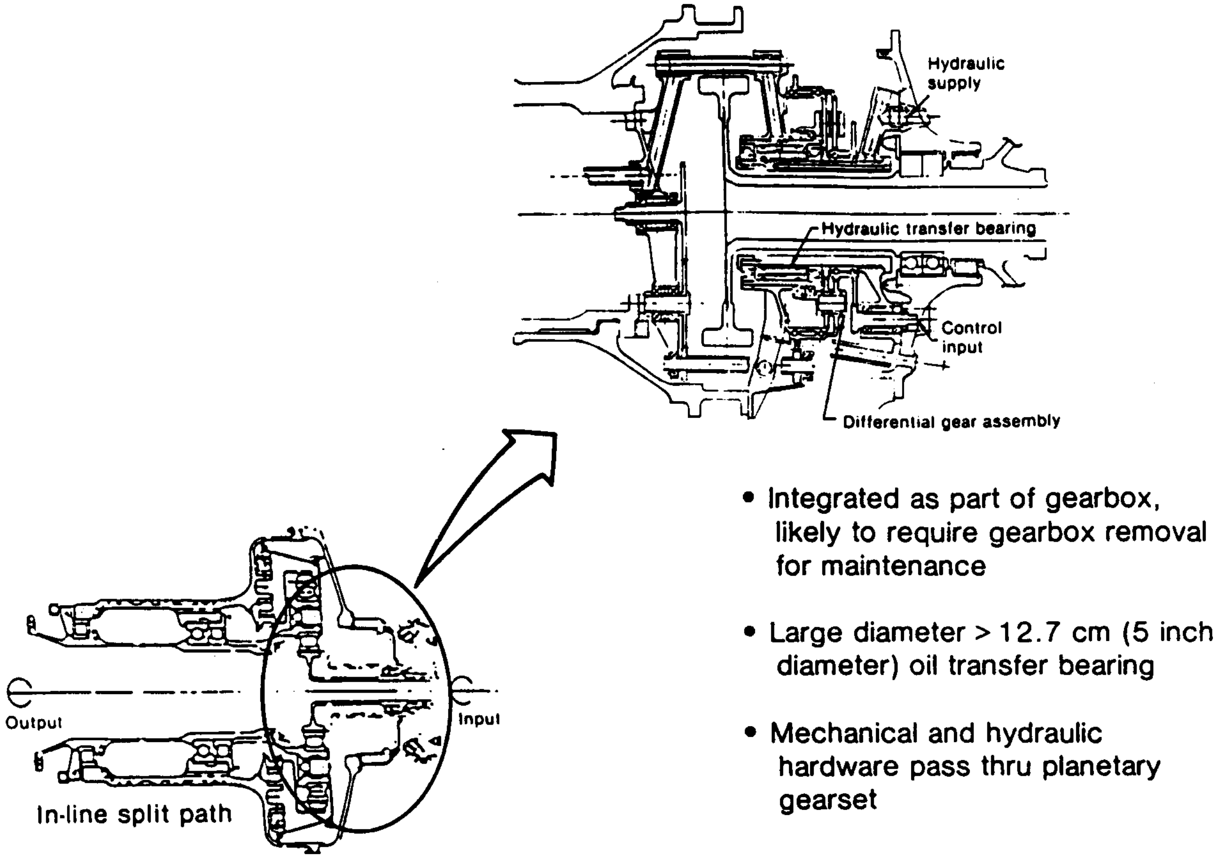
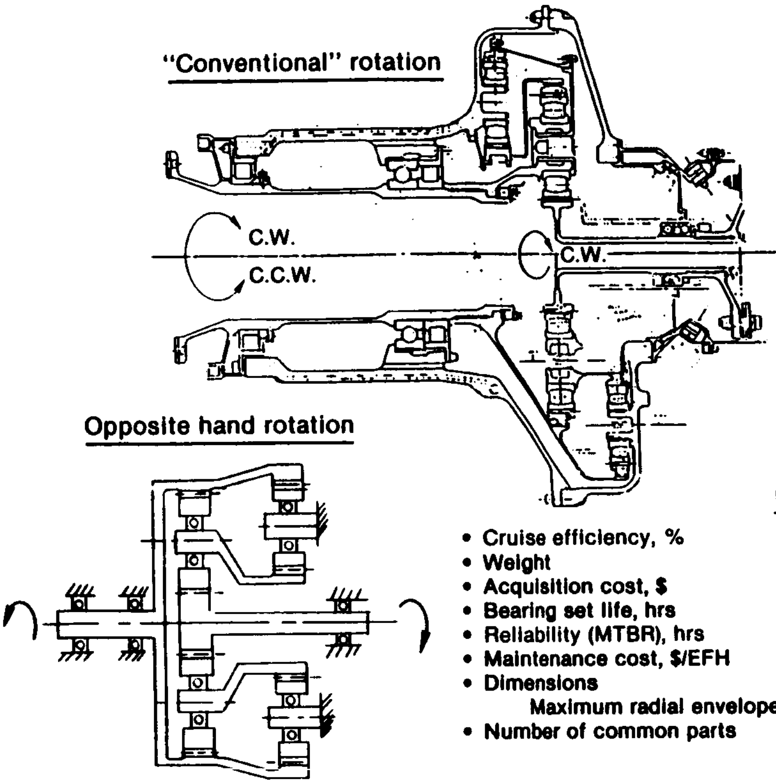


Figure 4.3-5 In-Line Gearbox/Pitch Control Integration Highlights - With an in-line reduction gear system, the pitch control is integrated with the gearbox; thus, the gearbox must be removed for pitch control maintenance actions. (J27638-93)

Input from the aircraft manufacturers indicated that an opposite hand rotation Prop-Fan may be required to optimize wing interference and wing aerodynamics, and to reduce noise. Figure 4.3-6 shows the impact of opposite rotation on the in-line split path gearbox concept. A schematic of the opposite hand rotation configuration is shown in the lower left hand portion of the figure. Very few of the parts in the opposite hand rotation gearbox are interchangeable with the parts in the conventional rotation gearbox. To implement the opposite rotation, the carrier of the first stage planetary gear set must be connected to the spur gear of the second stage, resulting in a 70% increase in the planetary bearing speeds. The higher speed produces a significant increase in the bearing centrifugal loads. This reduces the bearing set life by approximately 20% and increases the maintenance cost by about 5%. Prop-Fan pitch control accessibility is more difficult because the second stage must be moved aft of the first stage gearing. Since the first stage pinion gearset speed is 70% higher than the Prop-Fan output speed, an additional transfer bearing is required to accommodate the higher first stage speed; this increases the complexity of the pitch control.



	Base design	Modified for opposite rotation
• Cruise efficiency, %	Base	~Base
• Weight	Base	~Base
• Acquisition cost, $	Base	~Base
• Bearing set life, hrs	Base	−20%
• Reliability (MTBR), hrs	15,000	15,000
• Maintenance cost, $/EFH	Base	+5%
• Dimensions		
Maximum radial envelope	Base	+10%
• Number of common parts	Base	Very few

Figure 4.3-6 Impact of Opposite Hand Rotation on the Single Rotation In-Line Split Path Reduction Gear Concept - An opposite hand rotation Prop-Fan will optimize wing interference and aerodynamics and reduce noise. (J27638-87)

Offset Compound Idler Reduction Gear Configuration

The offset compound idler reduction gear concept is illustrated in Figure 4.3-7. A schematic of the gearbox is presented on the left side of the figure. The total gear (speed) reduction is 9.6, using a total of six gears and twelve bearings. The inlet pinion gear drives two idler gears for the first stage speed reduction of 3.1. Each of the two idler gears is integrated with a co-axial pinion and together drive a gear attached to the output shaft of the second stage with a speed reduction ratio of 3.1. Six gears and twelve bearings are used for the concept.

Since the total input power from the engine is divided between the two idler gears, the input load (or power) must be shared equally. Four load sharing techniques were investigated: (1) a vertical floating pinion, (2) an axial floating pinion system, (3) an external mechanical balance beam arrangement, and (4) a hydraulic thrust piston arrangement. The vertical floating pinion was discounted because large gears were required to meet the offset installation requirements. The axial floating pinion system required additional axial

Conventional rotation

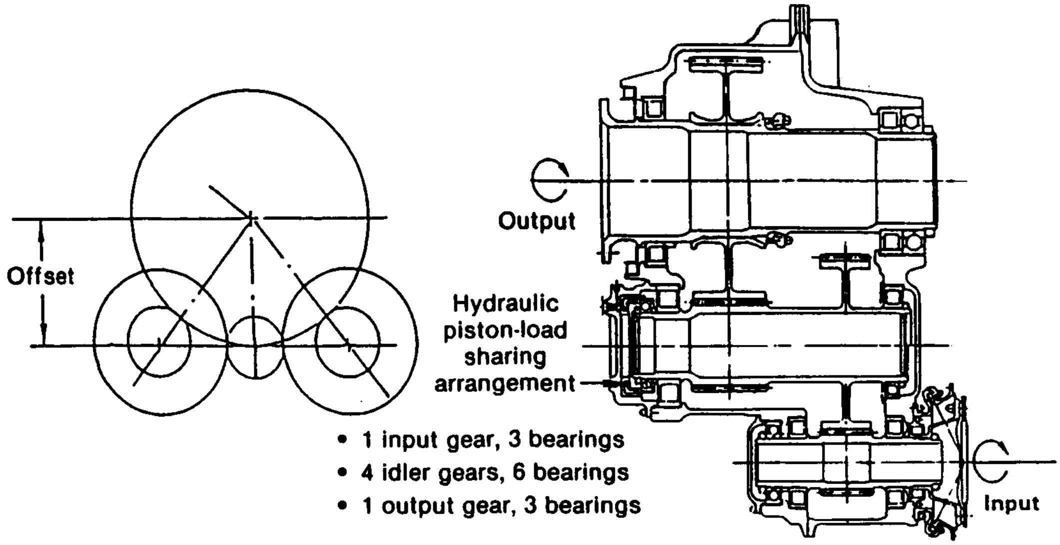


Figure 4.3-7 Offset Compound Idler Reduction Gear Concept - This system provides a total speed reduction of 9.6 and features a modular design. (J27638-88)

space within the gearbox as well as additional gears and would have increased the weight of the gearbox 5 to 10%. The external mechanical balance beam required a 15 - 20% increase in axial space, making it a very cumbersome system. The hydraulic thrust piston arrangement was selected because it had a minimal impact on gearbox weight and installation complexity. The hydraulic thrust piston load sharing concept is illustrated in the gearbox cross section (Figure 4.3-7).

As with the in-line split path reduction gear concept, the design includes an aircraft accessory pad, a Prop-Fan brake, and subassemblies which can be removed for inspection and/or repair without removing the entire gearbox from the aircraft. The removable subassemblies are listed in Table 4.3-V. The offset reduction gear concept provides all of the modular features of the in-line split path concept. In addition, the Prop-Fan pitch control can be removed for inspection or repair without removing the gearbox from the aircraft.

Gear stresses for the offset compound idler gearbox concept are summarized in Table 4.3-VIII. The limiting gear stress level, used to size the gears, is based on the bending unidirectional tooth loading. Gear size is minimized by using a high contact ratio gear tooth design. The design evaluation resulted in setting the second gear stage stresses closer to the design limits than the first stage stresses because the gear weight is concentrated in the second

stage. The gear dimensions for each stage are set by considerations of relative speed ratio, center distance, and pinion gear face width-to-diameter ratio. By reducing the first stage gear face width-to-diameter ratio from 0.85 to 0.73, Hertz and bending stresses could be increased to the design limits. This was judged to reduce gear weight by less than 5%; therefore further design iterations were not performed.

TABLE 4.3-VIII
GEAR STRESS SUMMARY OFFSET COMPOUND IDLER GEARBOX
(Conventional Rotation)

Gear		Stress		
Stage	Shaft	Type	Value MPa (psi)	Allowable MPa (psi)
First	Input	Hertz	894 (129,600)	1041 (151,000)
		Bending	348 (50,400)	414 (60,000)*
	Idler	Hertz	894 (129,600)	1041 (151,000)
		Bending	314 (45,500)	414 (60,000)*
Second	Idler	Hertz	1035 (150,000)	1041 (151,000)
		Bending	428 (60,000)	414 (60,000)*
	Output	Hertz	1035 (150,100)	1041 (151,000)
		Bending	359 (52,100)	414 (60,000)*

*Bending Limit Based on Unidirectional Tooth Loading

Table 4.3-IX summarizes the individual bearing lives required to meet the total bearing system design objective of a B10 life of 18,000 hours. As seen in the table, several of the individual bearing lives are in the 50,000 to 100,000 hour range. These individual bearing lives are lower than the bearing lives for the in-line gearbox because the number of bearings is reduced from 21 to 12. Several ball bearings have lives of 300,000 hours or more because bearing sizes are determined by shaft and housing requirements rather than bearing load.

The impact of opposite hand rotation on the offset compound idler reduction gear configuration is shown in Figure 4.3-8. Two idler gears and four bearings are added to the conventional system to achieve opposite hand rotation. The idler gears used in the conventional rotation gearbox are moved outward approximately 2.5 cm (1.0 in) (in a 12,000 horsepower size gearbox) to disengage the output gearing. The additional parts required for opposite hand rotation reduce the efficiency and reliability of the gearbox, as well as increasing weight, cost, and maximum radial diameter. However, there are more common parts in the conventional and opposite rotation offset gearbox configurations than in the conventional and opposite rotation in-line gearbox configurations.

TABLE 4.3-IX
BEARING LIFE SUMMARY OFFSET COMPOUND IDLER GEARBOX
(Conventional Rotation)

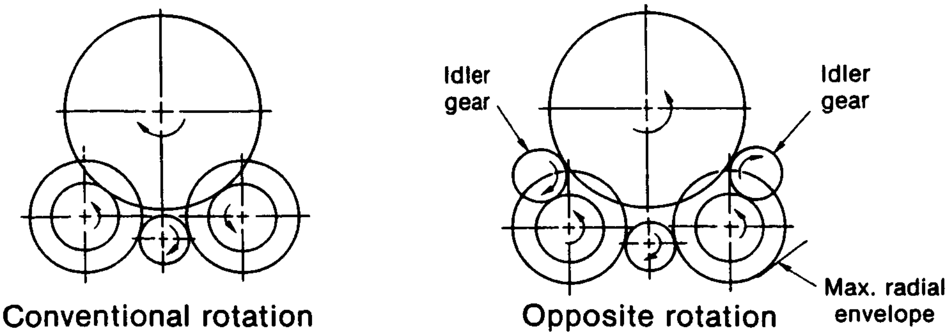
Bearing			Life, B10*
Location	Type	Quantity	Individual Bearing hours
Input Shaft	Roller	1	80,000
	Roller	1	98,000
	Ball	1	84,000
Left Idler	Roller	1	90,000
	Roller	1	108,000
	Ball	1	10^6
Right Idler	Roller	1	72,000
	Roller	1	48,000
	Ball	1	10^6
Output Shaft	Roller	1	100,000
	Roller	1	140,000
	Ball	1	300,000
		12	
Bearing System Life			18,800

*Equivalent life that 90% of all bearing sets will meet or exceed

Pitch control integration for the offset gearbox is highlighted in Figure 4.3-9. As illustrated in the sketch in the lower left hand corner, the pitch control is readily accessible from the rear of the gearbox, permitting pitch control maintenance actions "on the wing" of the aircraft. The oil transfer bearing is smaller than the bearing required for the in-line gearbox concept. This smaller bearing provides the potential for reduced weight as well as a simplified design for transferring the 20,684 to 34,474 Pa (3000 to 5000 psi) fluid used for Prop-Fan blade pitch change.

Pitch Control Comparison

The characteristics of the pitch control systems for the in-line and offset reduction gear configurations are summarized in Figure 4.3-10. The sketches shown at the top of the figure are roughly to scale. The slip ring assembly for electrical input to the Prop-Fan is readily located at the rear end of the offset gearbox pitch control system. However, for the in-line gearbox system, the slip ring assembly is placed at the front of the gearbox. The impact of the hydraulic transfer bearing, mechanical input, and maintenance of the in-line and offset gearbox concepts are highlighted in the figure.



	Base design	Modified for opposite rotation
• Cruise efficiency, %	Base	−0.3
• Weight	Base	+18%
• Acquisition cost, $	Base	+18%
• Bearing set life, hrs	Base	Base
• Reliability (MTBR), hrs	33,000	27,300
• Maintenance cost, $/EFH	Base	+15%
• Dimensions Max. radial envelope, ins.	Base	+ 5%
• Number of common parts	Base	Large number

Figure 4.3-8 Impact of Opposite Hand Rotation on Offset Compound Idler Reduction Gear Configuration - Opposite hand rotation reduces the efficiency and reliability of the gearbox and increases weight and cost. (J27638-61)

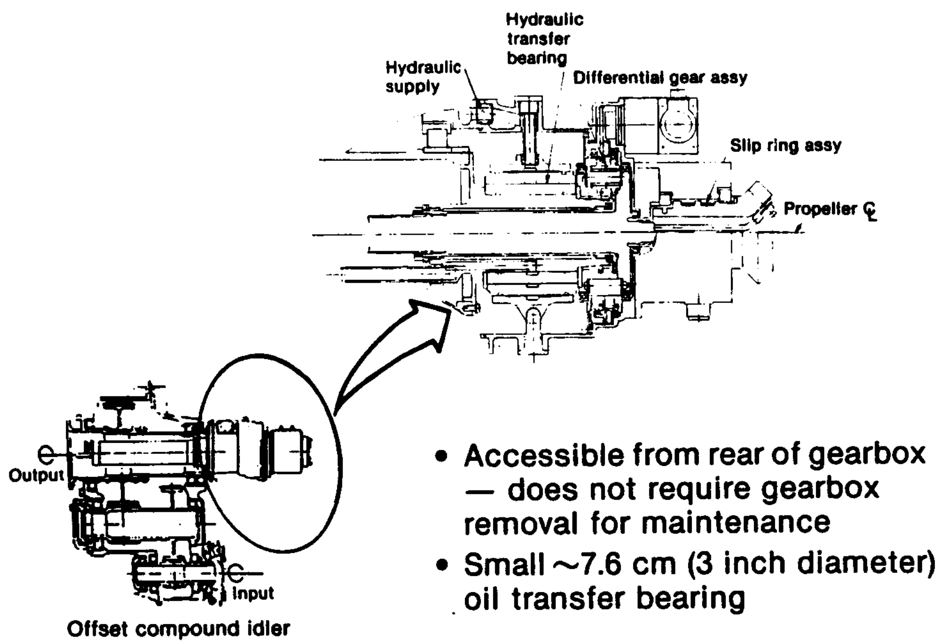
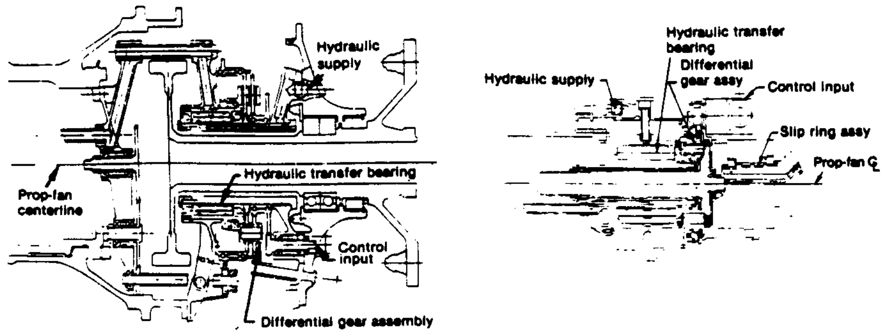


Figure 4.3-9 Offset Gearbox/Pitch Control Integration - With an offset reduction gear configuration, pitch control maintenance actions can be performed on the wing of the aircraft. (J27638-94)

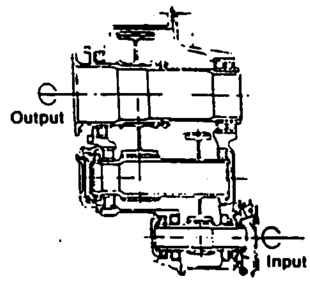


	In-Line	Offset
• MECHANICAL AND HYDRAULIC INPUT	• Through gear train	• Through prop ℄ transfer tube
• ELECTRICAL INPUT	• Through slip ring on rotor	• Through prop ℄ transfer tube
• TRANSFER BEARING	• Large diameter	• Small diameter
• MECHANICAL INPUT (DIFFERENTIAL GEAR TRAIN)	• Right angle drive	• Small and simple
• MAINTENANCE	• Removal of Prop-Fan assembly	• Modular and accessible for maintenance

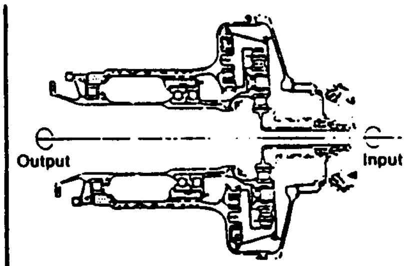
Figure 4.3-10 Comparison of Pitch Control Systems for In-Line and Offset Reduction Gear Configurations - The pitch control for the offset gearbox is more accessible than the pitch control for the in-line gearbox. (J27638-95)

In-Line/Offset Gearbox Configuration Comparison

The significant characteristics of the in-line split path and offset compound idler reduction gear configurations are compared in Figure 4.3-11. The advantages of the offset compound idler system include: greater efficiency, less major parts (bearings and gears), lower acquisition and maintenance cost, greater reliability, and more effective pitch control integration. The in-line split path gearbox configuration offers a significant weight advantage; in addition, it can result in a slimmer nacelle because of its smaller diameter. The final selection of a gearbox concept for a Prop-Fan propulsion system must be made in integrated studies with the airframe manufacturers. Therefore, both the in-line and offset reduction gear configurations have been included in the Propulsion System Integration Package.



Offset compound idler



In-line split path

	Offset compound idler		In-line split path	
• Rotation	Conventional	Opposite	Conventional	Opposite
• Efficiency	Base	−0.3%	−0.2%	−0.2%
• Number of bearings/gears	12/6	16/8	21/20	21/20
• Diameter	Base	+5%	−34%	−27%
• Weight	Base	+18%	−35%	−35%
• Acquisition cost	Base	+18%	+23%	+23%
• Reliability (MTBR, hrs)	32,700	27,300	15,000	15,000
• Maintenance cost, $/FH [1] (1982 $)	2.40	2.80	4.80	5.05
• Pitch control integration	Base		More difficult	

[1] In nominal 12,000 horsepower size

Figure 4.3-11 Comparison of In-Line and Offset Reduction Gear Configurations - Each configuration offers advantages and disadvantages; the final selection must be made in conjunction with the airframe manufacturers. (J27638-108)

When compared in a Prop-Fan powered aircraft, the significant weight reduction and slimmer nacelle of the in-line gearbox overcame the acquisition and maintenance cost advantage of the offset gearbox. Results of the fuel burned and direct operating cost evaluation are summarized below.

Gearbox	Rotation*	in Fuel Burn, %	Direct Operating Cost, %
Offset Compound Idler	Opposite	Base	Base
Offset Compound Idler	Same	-0.4	-0.25
In-Line Split Path	Opposite	-1.4	-0.4
In-Line Split Path	Same	-1.4	-0.4

* Propeller direction of rotation, left vs right engines

Technical Considerations Requiring Further Study

The major technical consideration requiring study beyond the scope of the current contract is summarized in Table 4.3.-X. Preliminary design studies should be conducted to identify innovative approaches to lowering the acquisition cost and improving the modularity of the in-line reduction gear system through improved accessibility to the Prop-Fan pitch conrol.

TABLE 4.3-X
REDUCTION GEAR/PITCH CONTROL INTEGRATION
TECHNICAL CONSIDERATION REQUIRING ADDITIONAL EFFORT

- In-Line Gearbox/Pitch Control Integration and Modularity

4.3.2.3 Engine and Aircraft Accessory Locations for Power Extraction

Various mounting locations for engine and aircraft related accessories were evaluated. Engine related accessories, which include the fuel pump, electronic controls for the propulsion system, lubrication pumps, electrical generator and starter, will be powered from the high-pressure spool through a gear drive system. Aircraft related accessories, such as hydraulic pumps and integrated drive electrical generators, can either be mounted on the engine or on the Prop-Fan reduction gearbox.

The airframe manufacturers indicated that sufficient data are not yet available to define the optimum location for the aircraft related accessories. Therefore, options for both engine and aircraft mounting locations have been included in the Propulsion System Integration Package. Preliminary studies would appear to favor engine mounted accessories for an in-line gearbox installation and gearbox mounted accessories for an offset reduction gear installation. In the in-line installation, inlet ducting is crowded around the gearbox; thus, panels would have to be included in the nacelle to provide access to gearbox-mounted airframe accessories. The engine mounting location alleviates this problem, as indicated in Figure 4.3-12. In the offset installation, the limited space available around the engine makes the gearbox mounting location more attractive, as indicated in Figure 4.3-13.

The final selection of an accessory mounting location will require trade studies with the airframe manufacturers. The key issues which must be addressed are summarized in Table 4.3-XI.

TABLE 4.3-XI
ENGINE AND AIRCRAFT ACCESSORY TECHNICAL CONSIDERATIONS
REQUIRING ADDITIONAL EFFORT

- Conduct trade studies with airframe manufacturers to determine proper location of airframe accessories
- Conduct trade studies with airframe manufacturers to evaluate "all-electric" accessory technology

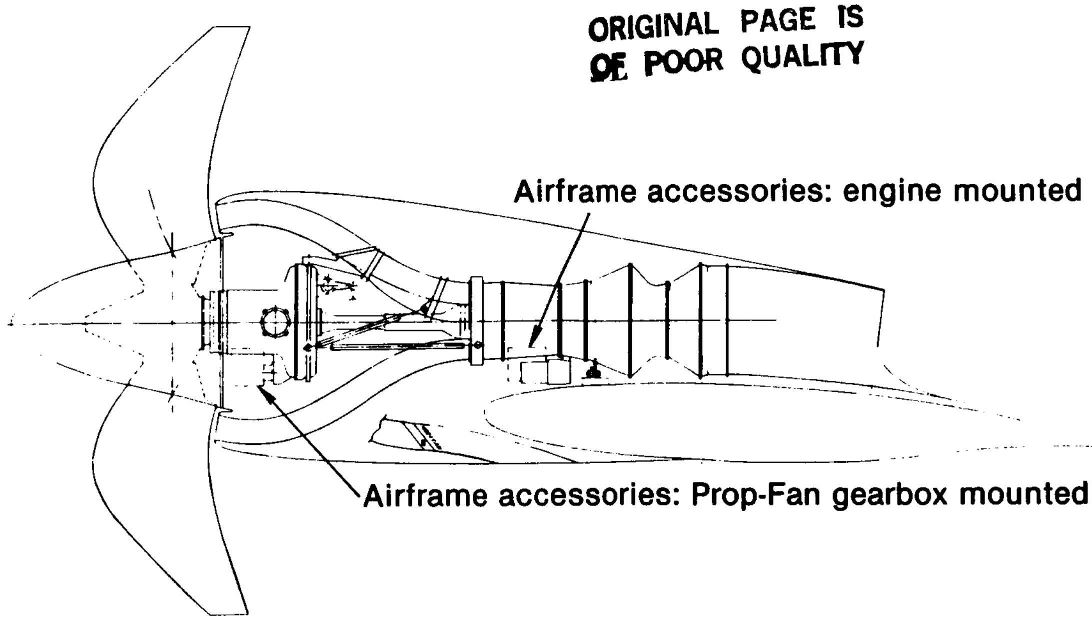


Figure 4.3-12 Aircraft Accessory Mounting Options for an In-Line Gearbox Installation - Two mounting options are provided for location of aircraft accessories. (J27638-8)

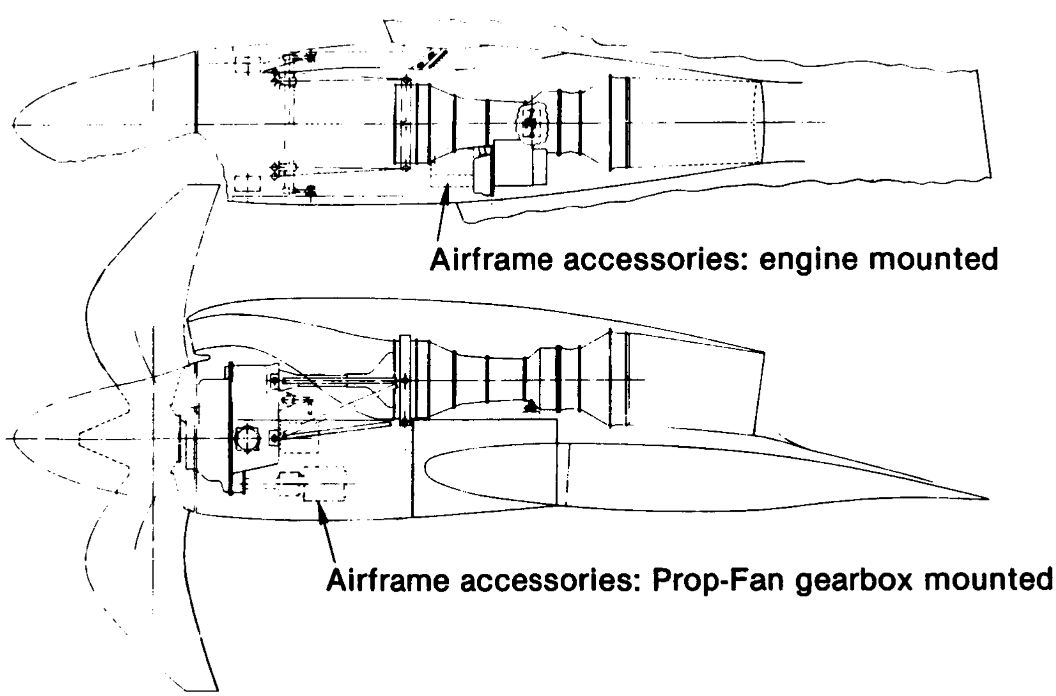


Figure 4.3-13 Aircraft Accessory Mounting Options for an Offset Gearbox Installation - Two mounting options are provided for location of aircraft accessories. (J27638-11)

4.3.2.4 Inlet Configurations

Since the interaction between the Prop-Fan and the inlet is a key factor in designing an efficient turboprop propulsion system, several inlet configurations were evaluated in the APET Program. Annular, trifurcated, bifurcated and chin inlet concepts were considered for an in-line gearbox installation. Chin and bifurcated inlets were viable candidates for an offset gearbox installation. After an evaluation of the inlets, summarized below, the chin inlet was selected for the offset gearbox installation and the bifurcated inlet was selected for the in-line gearbox installation. Both of these concepts have been included in the Propulsion System Integration Package.

The evaluation focused on major technical concerns for the inlet concepts, changes in inlet dimensions required for two-spool and three-spool engine configurations, and inlet pressure losses. Technical concerns included: the impact of Prop-Fan spinner boundary air on inlet performance, engine compressor airflow distortion, Prop-Fan back pressure effects, bird ingestion/dirt removal, angular airflow sensitivity, inlet anti-icing, and compatibility with the gearbox. A qualitative assessment of each inlet concept was made on the basis of these parameters. These judgements were confirmed by the airframe manufacturers. Technical issues requiring additional effort beyond the scope of the current contract were also identified.

Inlets for In-Line Gearbox Installations

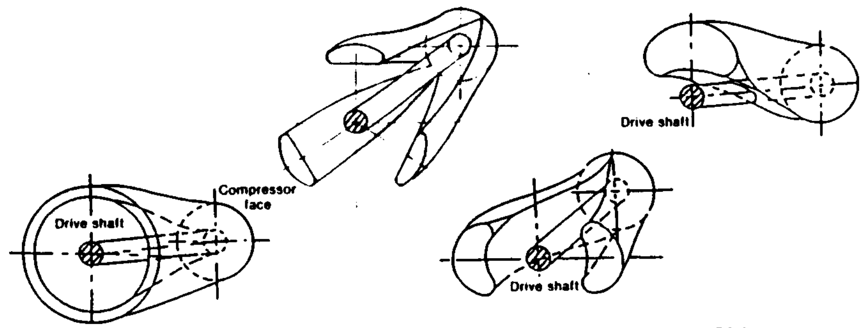
Technical Concerns - Annular, trifurcated, bifurcated and chin inlet concepts were considered for in-line gearbox installations. A qualitative assessment of the relative merits of each concept is presented in Table 4.3-XII. The results of the evaluation are discussed briefly.

With the annular inlet concept, pressure losses due to the spinner boundary layer are very significant. In all of the other configurations, the boundary layer generated by the spinner can be diverted away from the engine inlet, eliminating pressure losses. On the other hand, circumferential face distortion is less of a concern with the annular inlet concept than with any of the other inlet concepts evaluated.

Although the chin inlet concept produces the maximum back pressure distortion on the Prop-Fan, it is not considered a significant factor. Since bird ingestion and dirt removal are influenced primarily by inlet height, the chin inlet is also least favorable in this category. However, chin inlets continue to be widely used in current turboprop propulsion systems.

Increasing the number of inlet ducts increases the angular airflow sensitivity, making the trifurcated inlet least favorable in this category. Since the surface area requiring anti-icing is directly proportional to the inlet lip length, the chin inlet demonstrates an advantage. Finally, gearbox compatibility, which is related to geometric design considerations including the Prop-Fan and nacelle, favors the annular and trifurcated inlet concepts.

TABLE 4.3-XII
MAJOR TECHNICAL CONCERNS FOR INLETS USED WITH IN-LINE GEARBOX INSTALLATIONS



	Annular	Trifurcated	Bifurcated	Chin
• Spinner boundary layer loss	6.5% PT	0	0	0
• Distortion	Minimal	Medium	Medium	Maximum
• Prop back press.	Favorable	Less favorable	Less favorable	Least favorable
• Bird ingestion/dirt removal	Favorable	Favorable	Less favorable	Least favorable
• Angular airflow sensitivity	Favorable	Least favorable	Less favorable	Favorable
• Inlet anti-icing	Least favorable	Less favorable	Less favorable	Favorable
• Gearbox compatability	Favorable	Favorable	Less favorable	Least favorable

The impact of the various inlet configurations on (1) the distance between the engine inlet compressor face and inlet highlight plane and (2) the maximum inlet throat height for an in-line gearbox installation is summarized in Table 4.3-XIII. The in-line gearbox installation has the "slimmest" nacelle (0.28 nacelle diameter to Prop-Fan diameter ratio). Inlet length decreases with increasing number of inlets for all the concepts considered when the airflow turning requirement is met. For the bifurcated inlet concept, the inlet length is not significantly affected by the geometry of the compressor inlet in either the STS678 (two-spool engine) or the STS679 (three-spool engine) configuration. However, there is a 7 to 15 cm (3 to 6 in) change for the other inlet concepts evaluated.

The very small inlet throat height (4 cm (1.6 inches)) for the annular concept is approximately the height of the boundary layer of the airflow after it has passed over the Prop-Fan spinner. This results in the very high pressure loss shown in Table 4.3-XII.

Inlets for Offset Gearbox Installations

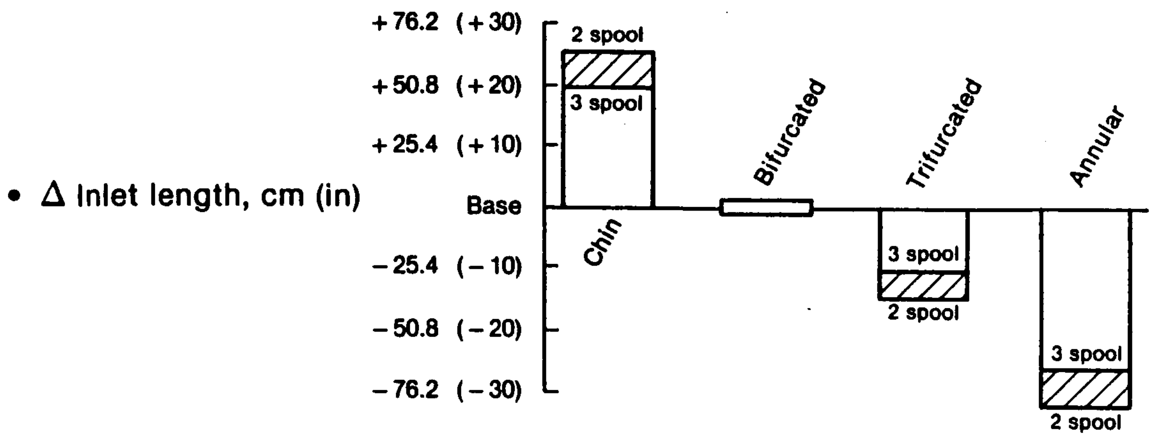
Bifurcated and chin inlet concepts were considered for an offset gearbox installation. Technical concerns are highlighted in Table 4.3-XIV; inlet dimensions are compared in Table 4.3-XV. A significant increase in length is required for the bifurcated inlet to meet the airflow turning requirements. As noted, the length of the chin inlet must be increased (15 cm (6 in)) to ensure compatibility with the STS678 two-spool engine.

TABLE 4.3-XIII
DIMENSIONAL COMPARISON OF INLETS FOR IN-LINE GEARBOX INSTALLATIONS

28% Nacelle
Inlet throat area/area
of compressor face = 1.0

Inlet throat area = 16.0 cm^2 (248 in^2)
Aspect ratio of inlet = 3

- Max. inlet throat height, cm (in) 24.13 (9.5) 17.02 (6.7) 13.97 (5.5) 4.06 (1.6)



Inlet Pressure Losses

The estimated inlet pressure losses for the candidate inlet concepts are presented in Table 4.3-XVI. The assessment of pressure losses was determined by considering the total inlet duct surface area and boundary layer effects. Except for the annular inlet, which has spinner boundary layer losses, the pressure losses are not significantly different for any of the inlet concepts evaluated.

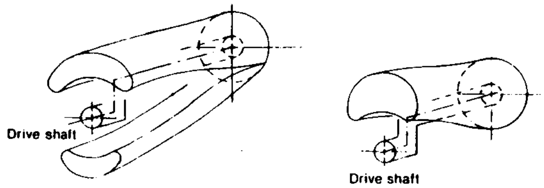
Inlet Selection

Based on these findings, the qualitative assessment of the inlet concepts presented in Tables 4.3-XI through 4.3-XVI, and comments from the airframe manufacturers, the chin inlet was selected for the offset gearbox installation and the bifurcated inlet for the in-line gearbox installation. Both concepts have been included in the Propulsion System Integration Package.

Technical Considerations Requiring Additional Study

While specific inlets have been selected for the in-line and offset reduction gear configurations, several technical considerations were identified which require additional analytical and/or experimental effort beyond the scope of the current contract. These issues are highlighted in Table 4.3-XVII. Many of these issues are covered in the Prop-Fan/Nacelle/Inlet/Compressor Technology Verification Plan presented in Section 4.5.

TABLE 4.3-XIV
MAJOR TECHNICAL CONCERNS FOR INLETS USED WITH IN-LINE GEARBOX INSTALLATIONS

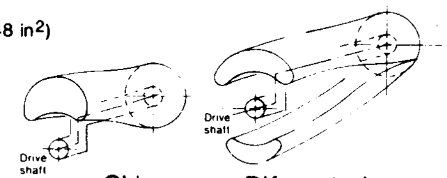


	Bifurcated	Chin
• Distortion	Minimal	Medium
• Prop back press.	Favorable	Less favorable
• Bird ingestion/dirt removal	Favorable	Less favorable
• Angular airflow sensitivity	Less favorable	Favorable
• Inlet anti-icing	Less favorable	Favorable
• Gearbox compatability	Favorable	Favorable

TABLE 4.3-XV
DIMENSIONAL COMPARISON OF INLETS FOR OFFSET GEARBOX INSTALLATIONS

32% Nacelle
Inlet throat area/area of compressor face = 1.0
Inlet throat area = 15.5 cm^2 (248 in^2)

Aspect ratio of inlet = 3
Offset = 45.72 cm (18 inches)



	Chin	Bifurcated
• Max. inlet throat height, cm (in)	24.13 (9.5)	17.02 (6.7)

• Δ Inlet length, cm (in)

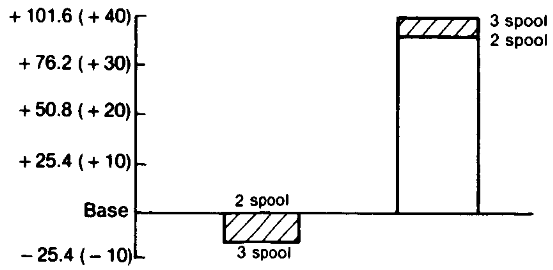


TABLE 4.3-XVI
ESTIMATED INLET PRESSURE LOSSES

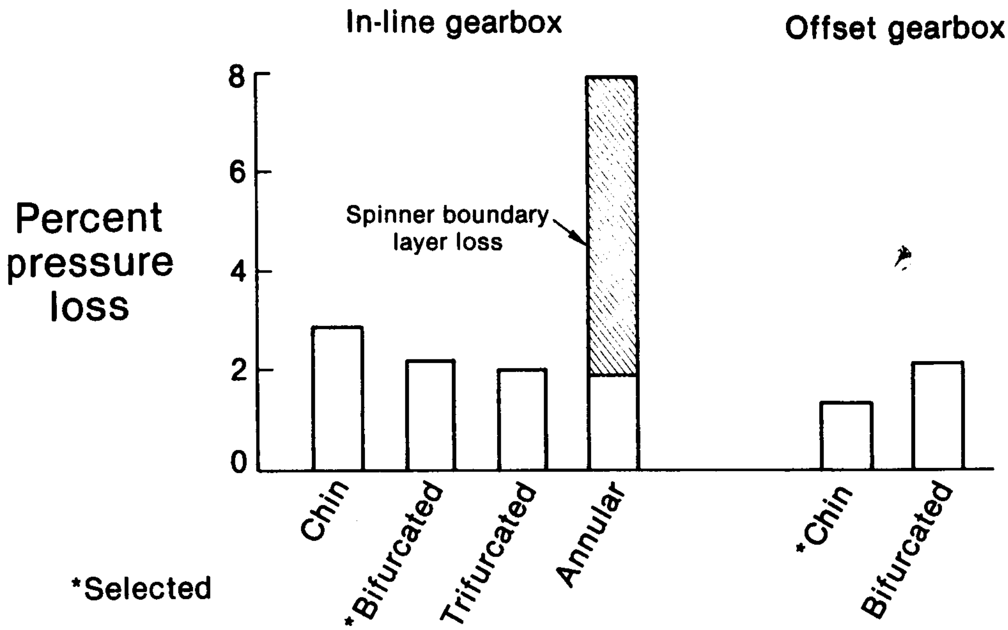


TABLE 4.3-XVII
INLET/NACELLE TECHNICAL CONSIDERATIONS REQUIRING ADDITIONAL STUDY EFFORT

- o Coupled Internal/External Analysis with Propeller Influence
- o Boundary Layer Removal Design System Development (Analysis and Empirical Efforts Required)
- o Nacelle External Design System Including:
  - Prop-Fan Profile and Swirl Effects
  - Prop-Fan Power Coefficient Effects
- o Internal Shaft Cover/Engine Compressor Inlet Shaft Fairing Design
- o Define Specific Inlet Anti-Icing Requirements and Patented Solutions
- o Define Specific Bird Ingestion and Engine Flowpath Dirt Removal Requirements and Potential Solutions

4.3.2.5 Oil Cooler Arrangements

An effective gearbox cooling and lubrication system ensures that the Prop-Fan reduction gear operates efficiently at maximum power conditions and helps in providing long life for the individual gears and bearings. During gearbox operation, power is lost due to sliding and rolling contact in the gear meshes, bearing rolling contact, windage effects, and oil churning. These losses become more critical at high horsepower and speed, significantly increasing the temperature of the oil used to lubricate the gearbox. An oil cooling system must be provided to dissipate the heat and prevent the oil from breaking down. At reduced power conditions, such as cruise operation, the full flow of oil is not required: gearbox efficiency can be improved by reducing the oil flow, thus eliminating unnecessary oil churning losses. A two-stage lubrication system, in which oil flow is reduced during cruise and below-cruise power conditions, satisfies both of these requirements.

To explore a variety of alternatives, five air/oil cooling concepts and one fuel/oil cooling concept were evaluated. Based on recommendations from the airframe manufacturers, a fuel/oil system, using fuel from the aircraft tanks as a heat sink, with a supplementary air/oil cooler for auxiliary usage, was selected for the Propulsion System Integration Package.

The operating characteristics of a two-stage lubrication system for a 12,000 horsepower size reduction gear are shown in Figure 4.3-14. The data are based on the offset compound idler gearbox concept which has an efficiency of 99% at the maximum power (takeoff) condition using a modular (two-stage) lubrication system. Lines of constant efficiency have been included for comparison. The figure shows how oil flow is modulated to ensure maximum efficiency at each operating condition.

Table 4.3-XVIII presents typical gearbox oil heat rejection rates at discrete operating conditions throughout the flight cycle. (The values in this table correspond to the equivalent points in Figure 4.3-14.) Using this data, a plate-fin air/oil heat exchanger was designed which would effectively dissipate the heat in the 12,000 horsepower size Prop-Fan reduction gear. The characteristics of this heat exchanger are described in Table 4.3-XIX. The critical sizing point for the heat exchanger is low aircraft speed or static operating conditions, which dictates a low air-side pressure drop configuration. At other operating conditions in the flight cycle, ample pressure is provided by aircraft ram effects and airflow is controlled by valves or variable geometry.

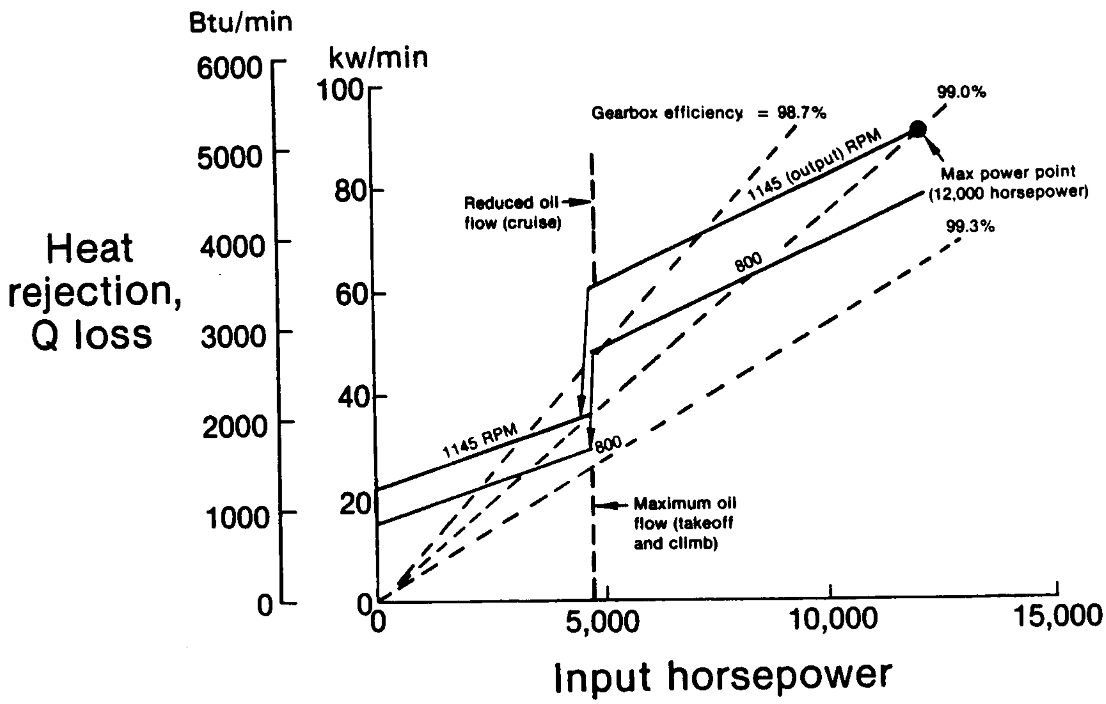


Figure 4.3-14 Heat Rejection Data for a 12,000 Horsepower Size Reduction Gearbox - With a two-stage lubrication system, oil flow is modulated to ensure maximum efficiency at a variety of operating conditions. (J27638-15)

Using the plate-fin air/oil heat exchanger system, five air/oil cooler concepts and one fuel/oil cooler concept were evaluated. These concepts cover a variety of approaches to dissipating the heat in the gearbox oil, including variable valves, variable geometry, and use of the fuel in the aircraft tanks as a heat sink. A brief description of each system follows.

TABLE 4.3-XVIII
TYPICAL PROP-FAN ENGINE FLIGHT CYCLE AND
HEAT REJECTION FROM 12,000 HORSEPOWER SIZE GEARBOX - STANDARD DAY

Flight Condition	Altitude m (ft)	Mach Number	Power (Thrust)	Prop rpm	Gearbox Input Horsepower	Heat Rejection KW/Min (Btu/Min)	Time, Minutes At End Segment
Taxi	0 (0)	0	Idle	567	179	11.2 (636)	9.0
Takeoff	0 (0)	0	Takeoff	1145	10,809	85.5 (4863)	10.4
Climb	457 (1500)	0.39	Climb	1145	11,539	88.5 (5036)	10.4
Climb	3048 (10,000)	0.5	Climb	1145	10,309	83.4 (4745)	12.8
Climb	6096 (20,000)	0.6	Climb	1145	8896	77.5 (4410)	16.6
Climb	9144 (30,000)	0.74	Climb	1145	7233	70.6 (4016)	25.5
Climb	10,668 (35,000)	0.75	Climb	1145	6162	66.1 (3762)	31.7
Cruise (Begin)	10,668 (35,000)	0.75	Cruise	1145	4785	35.4 (2015)	31.7
Cruise (End)	10,668 (35,000)	0.75	Cruise	1145	4732	35.3 (2006)	52.1
Descent (Begin)	10,668 (35,000)	0.75	Flt. Idle	651	267	13.0 (741)	52.1
Descent (End)	457 (1,500)	0.39	Flt. Idle	627	257	12.5 (713)	68.9
Approach	0 (0)	0.2	30% Takeoff Thrust	900	2564	24.4 (1386)	72.1
Touchdown	0 (0)	0.18	Idle	586	196	11.6 (659)	72.2
Reverse	0 (0)	0.15	Approx. Takeoff	844	1874	21.3 (1212)	72.4
Taxi	0 (0)	0	Idle	567	179	11.2 (636)	77.4

TABLE 4.3-XIX
TYPICAL PLATE-FIN HEAT EXCHANGER CHARACTERISTICS FOR 12,000 HORSEPOWER SIZE GEARBOX

Dimensions

495 cm^2 (195 in^2) face area, 5.715 cm (2.25 in) thickness
Core and Header Weight (WET) = 15 kg (33 lb)

	SLTO	Cruise	Ground Idle
Oil Flow, kg/min (lb/min)	109 (240)	54 (120)	27 (59)
Oil In Temp., °C (°F)	121 (249)	139 (282)	132 (270)
Oil Out Temp., °C (°F)	99 (210)	121 (250)	121 (250)
Air Flow, kg/min (lb/min)	79 (175)	15 (32)	8 (16.6)
Air In Temp., °C (°F)	32 (90)	-5 (-16)	32 (90)
Air Out Temp., °C (°F)	96 (206)	121 (250)	121 (250)
Air Mach No. at Heat Exchanger Inlet	0.027	0.012	0.0025
Air Side Pressure Drop, P, MPa (psi)	2.21 (0.04)	0.965 (0.32)	0.275 (0.14)
Q, kw/min (Btu/min)	86 (4863)	35 (2015)	11 (636)

Air/Oil Cooler Concepts

Figure 4.3-15 shows an air/oil heat exchanger which has dual inlets with variable bypass valves. Dual inlets for the cooler are located downstream of the engine inlet to reduce and/or eliminate interference and interactions between the engine inlet and the cooler inlets. An ejector is used for flight conditions where there is insufficient pressure drop across the heat exchanger for effective heat dissipation. The cooler inlets incorporate flaps which are opened or closed at the proper times to eliminate secondary losses when the air/oil heat exchanger is not used to dissipate gearbox heat rejection.

Figure 4.3-16 illustrates a double flap concept which incorporates variable inlet and exhaust flaps to permit low profile drags at the 0.7 to 0.8 Mn cruise conditions. The takeoff operating condition is illustrated by dashed lines. The solid lines illustrate the partially open condition used to minimize drag during cruise operation. An ejector is incorporated to insure proper operation of the heat exchanger at conditions where there is insufficient pressure drop across the heat exchanger for effective gearbox heat dissipation. The system could use engine bleed air to induce airflow through the heat exchanger during ground and/or low speed aircraft flight conditions.

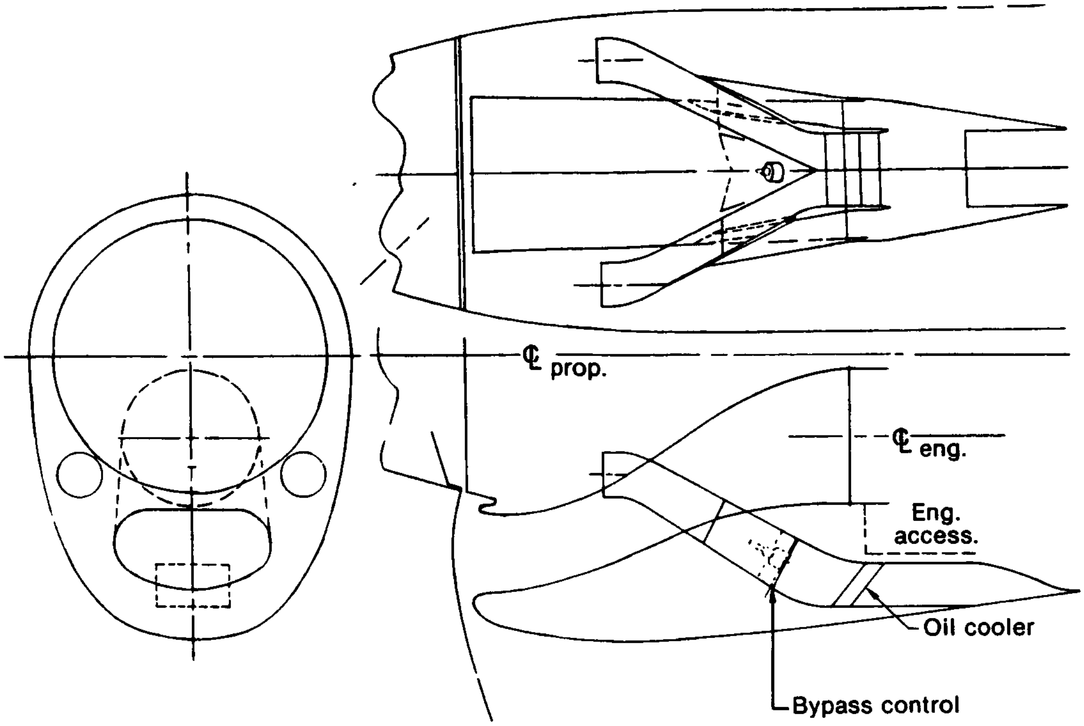


Figure 4.3-15 Air/Oil Heat Exchanger Concept; Dual Inlets with Variable Bypass Valves - In this concept, variable valves are used to regulate airflow. (J27638-89)

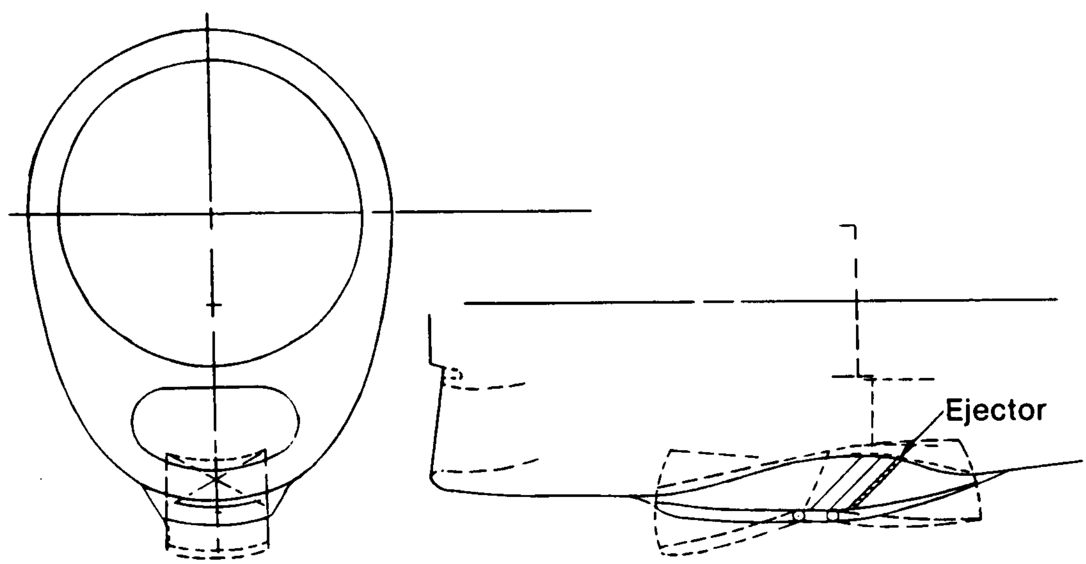


Figure 4.3-16 Double Flap Concept - In this concept, variable flaps are used to regulate airflow. (J27638-89A)

A variable cooler inlet concept similar to the double flap arrangement, (Figure 4.3-16) but using an alternate system to vary the inlet and exhaust areas, is illustrated in Figure 4.3-17. The inlet is made variable through a set of linkages which open and close the inlet area as required. As with the double flap concept, an ejector system is included.

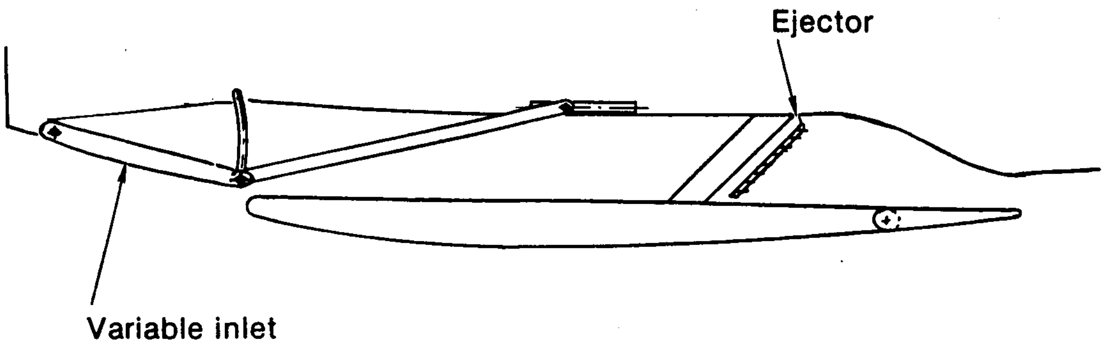


Figure 4.3-17 Variable Cooler Inlet Concept - This concept represents an alternate approach to variable inlet and exhaust geometry. (J27639-91)

Figure 4.3-18 shows an oil cooler concept using a common inlet with the engine at cruise. When maximum oil cooler airflow is required (takeoff and climb) a separate inlet opens to increase the airflow to the heat exchanger. This system would result in a low drag profile during cruise conditions. An ejector is also required to operate this system during ground and/or low speed aircraft flight conditions.

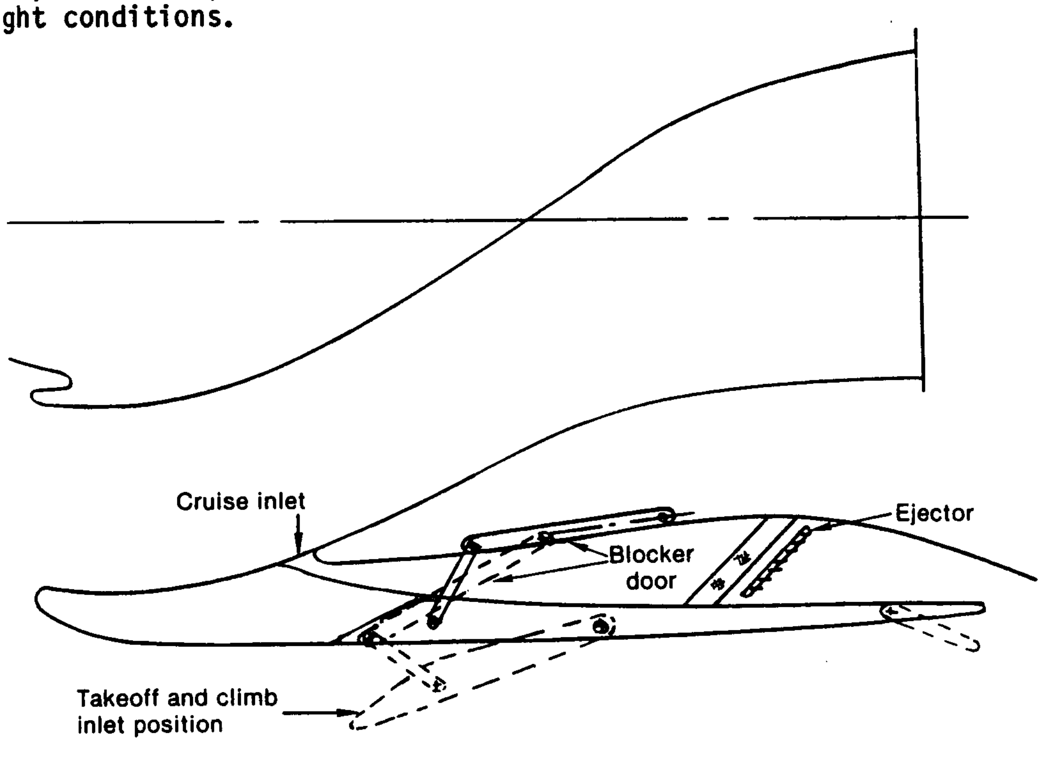


Figure 4.3-18 Oil Cooler Concept Using a Common Inlet at Cruise - This system presents a low drag profile at cruise conditions. (J27638-92)

In this air/oil heat exchanger concept, the engine inlet duct system is used for cooling as illustrated in Figure 4.3-19. The oil cooler is built into the wall of the inlet and rejects heat to the engine. This system insures airflow over the cooler at all flight conditions. However, the heated airflow increases turbine inlet temperature at takeoff and climb and results in a small cruise fuel consumption penalty. These trades, coupled with flight safety issues, must be evaluated in conjunction with studies conducted by the airframe manufacturers. The principal safety issue is a potential engine fire should an oil leak occur.

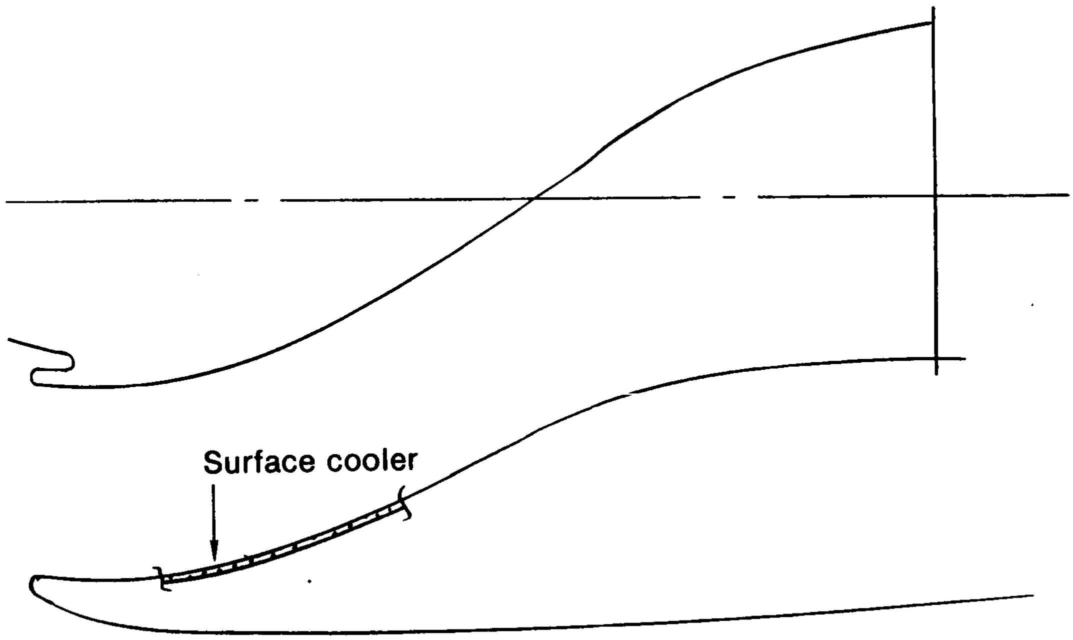


Figure 4.3-19 Inlet Duct Air/Oil Heat Exchanger Concept - This configuration insures airflow over the cooler at all flight conditions. (J27638-90)

Fuel/Oil Cooler Concept

The fuel being consumed by the highly fuel-efficient engines in the Prop-Fan propulsion system does not have a sufficient heat sink to dissipate the heat from the reduction gear. Thus, a heat rejection concept in which the fuel in the aircraft tanks is used as a heat sink was proposed by the airframe manufacturers (see Figure 4.3-20). This configuration eliminates the drag and reduces the inlet heating penalties associated with air/oil heat exchanger systems.

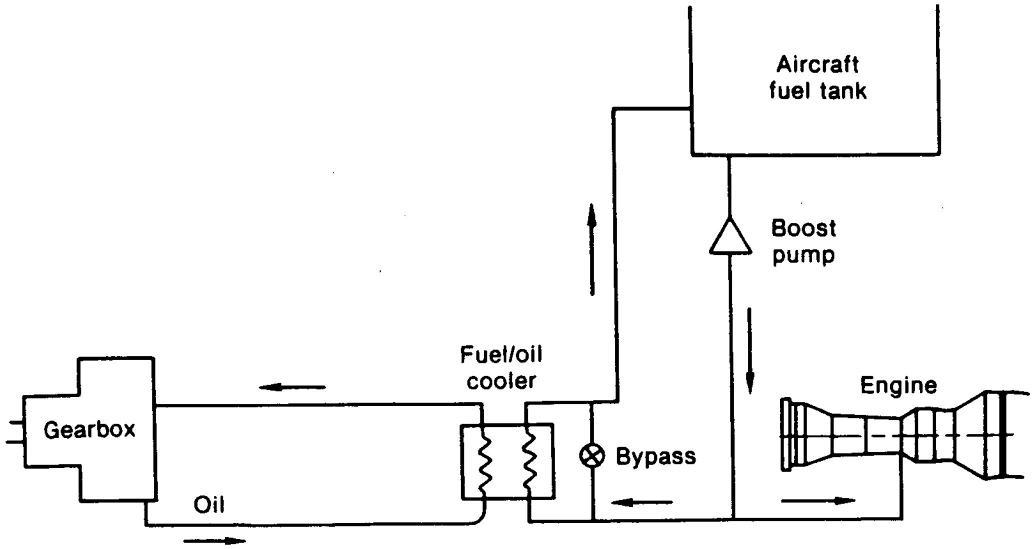


Figure 4.3-20 Fuel/Oil Cooler Concept - Heat from the gearbox is dissipated by using the fuel in the aircraft tank as a heat sink. (J27638-66A)

Based on comments from the airframe manufacturers, a supplementary air/oil cooler was added to the fuel/oil cooler concept, as shown in Figure 4.3-21. The supplementary cooler is used to dissipate gearbox heat rejection during operating conditions when fuel in the aircraft tanks is below the level required to absorb the gearbox heat rejection. Based on Pratt & Whitney's evaluation of the cooling concepts, as well as comments from the aircraft manufacturers, the fuel/oil cooler concept with a supplementary air/oil heat exchanger was selected for the Propulsion System Integration Package.

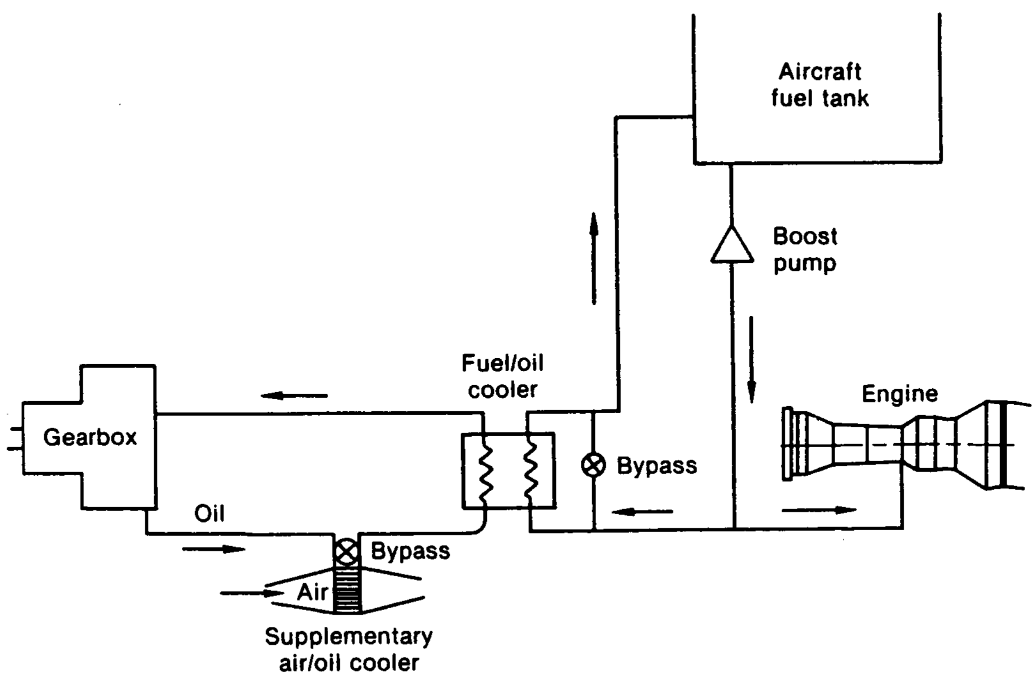


Figure 4.3-21 Fuel/Oil Cooling System with Supplementary Air/Oil Cooler - This system was selected for the Propulsion System Integration Package. (J27638-66)

Technical Considerations Requiring Additional Study

Technical considerations requiring study beyond the scope of the current contract were identified. These issues are summarized in Table 4.3-XX.

TABLE 4.3-XX
HEAT EXCHANGER TECHNICAL CONSIDERATIONS REQUIRING ADDITIONAL STUDY

- o Trade study of fuel/oil heat exchanger system with air/oil supplementary backup system for auxiliary aircraft operating conditions
- o Conceptual design studies with aircraft manufacturers to integrate engine, gearbox and other aircraft heat rejection subsystems
- o Conceptual design of supplementary air/oil cooler system (inlet, variable geometry, requirements, etc.) with airframe manufacturers

4.3.2.6 Propulsion System Control

The control system for the Prop-Fan propulsion system is an advanced design incorporating electronic circuitry, fiber optics, and dual redundancy in the vital control paths. This section describes the design approach, control modes, and items required to implement the control system. A detailed description of the system is provided, and technical issues requiring additional study effort are discussed.

Design Approach

The control system which will be designed for an advanced turboprop engine can employ many technological features currently under development. Control modes can be optimized to take advantage of the flexibility, compactness, and power of the computer. System communication can be enhanced by optics. Control redundancy can be optimized to achieve maximum reliability while minimizing weight and cost.

Electronics are being used with increasing frequency in aircraft control systems. The technology has been demonstrated in Full Authority Digital Electronic Controls (FADEC) on both military and commercial engines. The transition from mechanical systems has been aided by rapidly developing technology and the demonstrated reliability of electronic components and controls. As indicated in Table 4.3-XXI, full authority digital electronic controls ensure effective integration of the functional and performance requirements of the propulsion system. An electronic control system also provides an optimum balance between reliability, safety, maintainability, and cost. Considering these factors, and the other features highlighted in the table, a full authority digital electronic control system design was selected for the Prop-Fan powered aircraft.

TABLE 4.3-XXI
ADVANCED TURBOPROP CONTROL SYSTEM DESIGN APPROACH

- o Dual Channel Full Authority Digital Electronic Control (FADEC) to Control Prop-Fan and Engine is an Optimal Balance Between:
  - Integration of Functional and Performance Propulsion System Characteristics
  - Reliability
  - Safety
  - Maintainability
  - Acquisition Cost
- o Provision for Interfacing with Aircraft Flight Control System
- o Modular Construction and Simple Aircraft Installation/Removal
- o Integrated Total Propulsion System (Prop-Fan, Gearbox and Engine) Condition Monitoring System

The control system design will benefit from about 1.5 million flight hours of Pratt & Whitney and Hamilton Standard experience with engine mounted digital electronic controls. This background includes over one million flight hours with a digital supervisory control on the F100 military turbofan engine in F-15 and F-16 aircraft, and 300,000 hours of reliability demonstration testing on commercial Boeing 727 aircraft. The operational benefits of a dual-channel electronic control are demonstrated by the selection of this system for the PW2037 commercial transport engine which will power the Boeing 757, as well as the recently-introduced PW4000 family of engines. In other applications, requirements for safety-of-flight, reduced pilot workload, and a high level of aircraft availability (dispatchability) led to selection of the dual-channel configuration on a Boeing 747 as well as a Navy control technology program (FADEC) which verified the hardware and software used for a dual-channel arrangement.

Control Modes

There are several unique considerations in designing a control system for a Prop-Fan propulsion system; therefore, capabilities beyond those of current turbofan control systems are required. The variable pitch blade feature permits independent control of propeller speed and engine speed/power setting, increasing the complexity of the control system. Thrust control in both forward and reverse pitch also requires enhanced control system capabilities.

Advanced turboprop engines will be expected to incorporate the sophisticated thrust management capability found on the latest turbofan engines. The thrust setting system must be simple to operate and accurate in both forward and reverse modes. Devising adequate protective measures for limiting torque, preventing overspeed, and accommodating possible system faults will be complicated by the variable pitch propulsion system.

Electronic computation makes it possible to tailor propulsion system operation to the power setting regime, thus achieving maximum thrust at takeoff, low noise during approach, maximum thrust reversal effectiveness, and optimum fuel consumption during cruise. Integrating gas generator performance and Prop-Fan blade pitch setting offers additional flexibility in controlling transient operation during takeoff and landing conditions. Electronic computation also provides great flexibility in dealing with fault accommodation, leading to improved safety of flight. Major control mode features are summarized in Table 4.3-XXII.

TABLE 4.3-XXII
ADVANCED TURBOPROP CONTROL MODES

- Independent Control of Propeller (Synchrophasing, etc.) and Engine Speed/Power Setting
- Automatic Control in Steady State and Transient Operation for Forward and Reverse Thrust
- Protective Measures for Limiting Torque, Temperature, Overspeed, and Possible System Fault (Prop-Fan Feathering, Windmilling, etc.)

Implementation

Technology features required to implement a control system for a Prop-Fan powered aircraft are listed in Table 4.3-XXIII. The advanced turboprop control system will be based on the dual-channel, Full Authority Digital Electronic Control System currently under development for the PW2037 and PW4000 engines. Technology advances and related improvements in the reliability of electronics and optics will permit more selective redundancy in the control system. A reduction in component and circuit redundancy will reduce system cost and weight while maintaining the required reliability. Redundant electronic computation obviates the need for hydromechanical "backup" and the attendant cost and weight penalties, implementation compromises, and operational complications.

Use of advanced digital electronics and application of emerging technologies, including optic sensing and signal transmission concepts currently being developed in NASA programs will lead to a highly efficient, reliable control system. Optic technology can significantly reduce the electrical cabling required for a sophisticated multiple function control system. In the PW2037 engine control system, electrical cabling represents the third costliest and heaviest component in the control system.

TABLE 4.3-XXIII
ADVANCED TURBOPROP CONTROL SYSTEM IMPLEMENTATION

- Redundant Electronic Computation Power Supply Commands and Feedback
- Input and Output Electrical, Optical, and Pneumatic Conditioning
- Information From Control to Aircraft
  - Cockpit Instrumentation and Status Displays
  - Diagnostics and Condition Monitoring
- Optics Technology to Reduce Electrical Cabling and Immunity to External Electromagnetic Threats

Description of a Potential Turboprop Control System

The control system which has been designed for the Prop-Fan propulsion system is shown in block diagram form in Figure 4.3-22. The heart of the system is an electronic control unit housing circuitry for digital computation, input and output conditioning, and electrical power regulation. This circuitry gathers information from the propulsion system sensors and modulates the various engine functions including gas generator fuel flow, propeller pitch, variable compressor geometry, active clearance control (if required), and other performance optimization features.

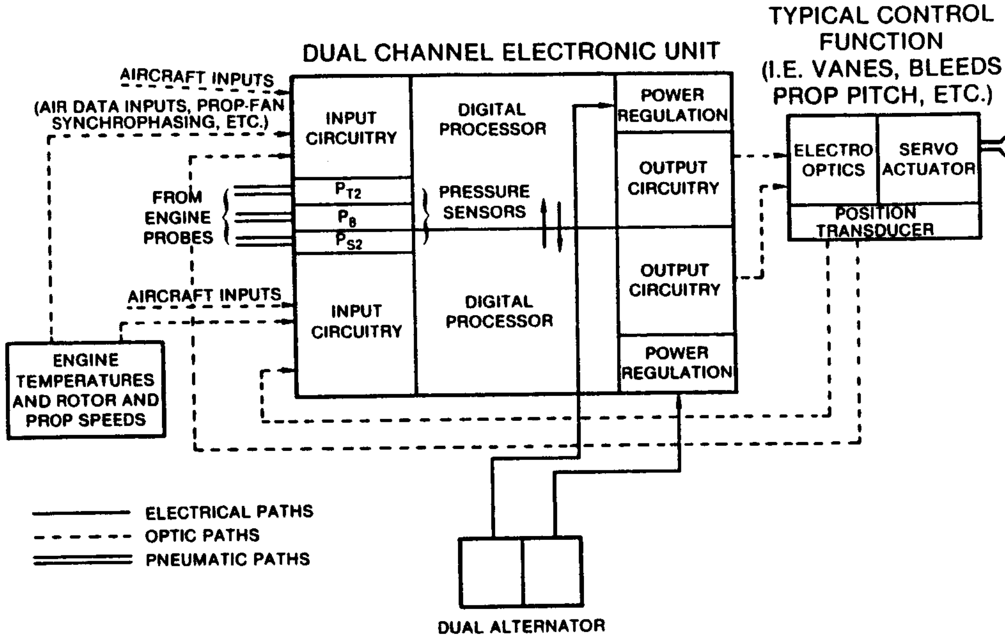


Figure 4.3-22 Advanced Turboprop Control System - This control system will use advanced technology to ensure maximum efficiency and reliability while reducing weight and cost. (J27638-3)

Dual redundancy is used in the vital control paths for power supply, computation commands, and feedback.

The electrical power required to operate the system will be provided by a dual winding permanent magnet alternator driven by the gas generator accessory drive system.

Aircraft information used by the control system will be provided via a digital fiber optic data bank feeding into each control computer channel. Information from the control system will be provided to the aircraft for instrumentation and status displays, diagnostics, etc. Feedback precision sensors and various control system actuators will be digitally compatible with optical transducers requiring no electrical excitation, similar to those being developed under

NASA contract NAS3-19898. Command signals from the control unit to the actuators can also be transmitted optically using technology being developed under NASA contracts NAS3-21809 and NAS3-22535. The fiber optics signal "conductors" reduce weight as much as four to five fold compared to current electrical units while requiring no electromagnetic shielding for interference or lightning induced effects. Other potential applications of optic technology include gas path temperature sensors which are being developed under NASA contract NAS3-21841.

While a full authority digital electronic control system has been designed for the Prop-Fan propulsion system, there are additional technical issues which must be addressed. These issues are summarized in Table 4.3-XXIV.

TABLE 4.3-XXIV
ADVANCED TURBOPROP ENGINE CONTROL TECHNICAL ISSUES REQUIRING ADDITIONAL ANALYTICAL EFFORTS

- Define Control Mode, Fault Logic, and Integrated Electronics for Candidate Engines to Improve Cost, Weight and Reliability
- Systems Integration with Aircraft for Optimal Use of All Electronic Aircraft Systems
- Optics Technology to Ensure Immunity to External Electromagnetic Threats (Lightning, etc.)

4.3.2.7 Prop-Fan Configuration

A ten-bladed Prop-Fan configuration was selected for the integrated propulsion system. The Prop-Fan has a diameter of 4.05 m (13.3 ft), and a loading of 34 shp/D^2 with a tip speed of 243 m/sec (800 ft/sec) at maximum climb conditions, 10,668 m (35,000 ft), 0.75 Mach number. The Prop-Fan diameter and loading were derived from a previous study conducted by Pratt & Whitney, the results of which are summarized in Figure 4.3-23. These results agree with independent studies conducted by the airframe manufacturers.

A mechanical description of the Prop-Fan, provided by Hamilton Standard, is presented in Figure 4.3-24. The interface with the gearbox, in this case an offset compound idler configuration, is shown in the figure. The Prop-Fan concept which is illustrated can also be used with an in-line gearbox configuration; however, the pitch control system would be quite different (see Section 4.3.1.2).

The Prop-Fan configuration which was selected is a conventional "tractor" installation. It is recommended that additional studies be conducted to evaluate a "pusher" Prop-Fan from the standpoint of performance, mounting, interface with the reduction gear, and aircraft installation.

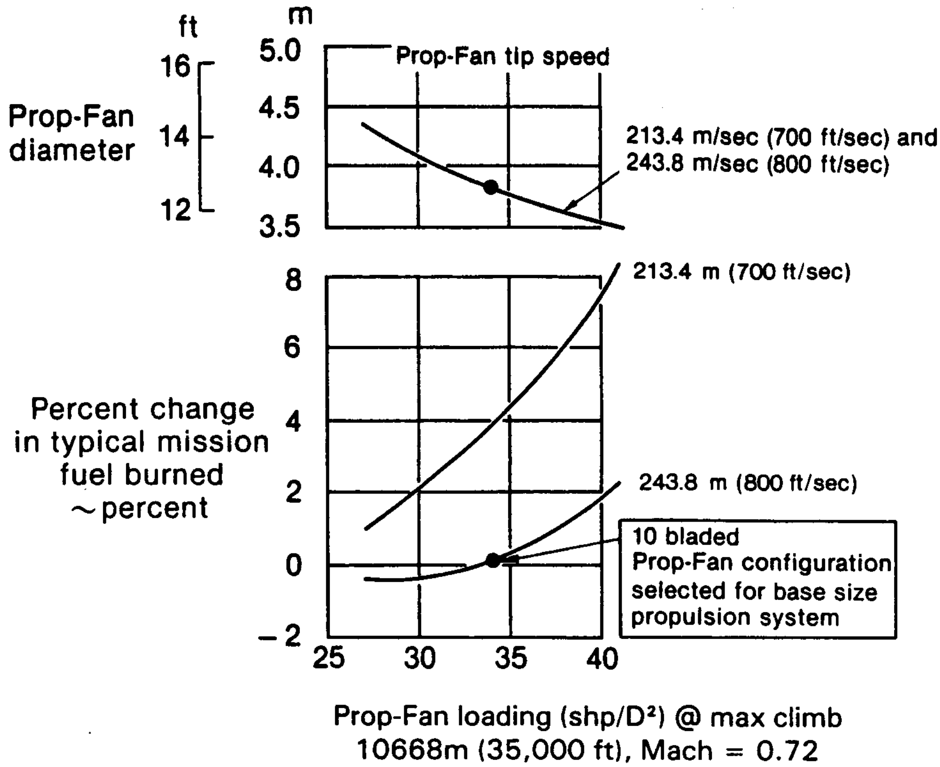


Figure 4.3-23 Prop-Fan Selection Trade Study - A ten-bladed Prop-Fan with a diameter of 4.05 m (13.3 ft) was selected for the Propulsion System Integration Package. (J27638-17)

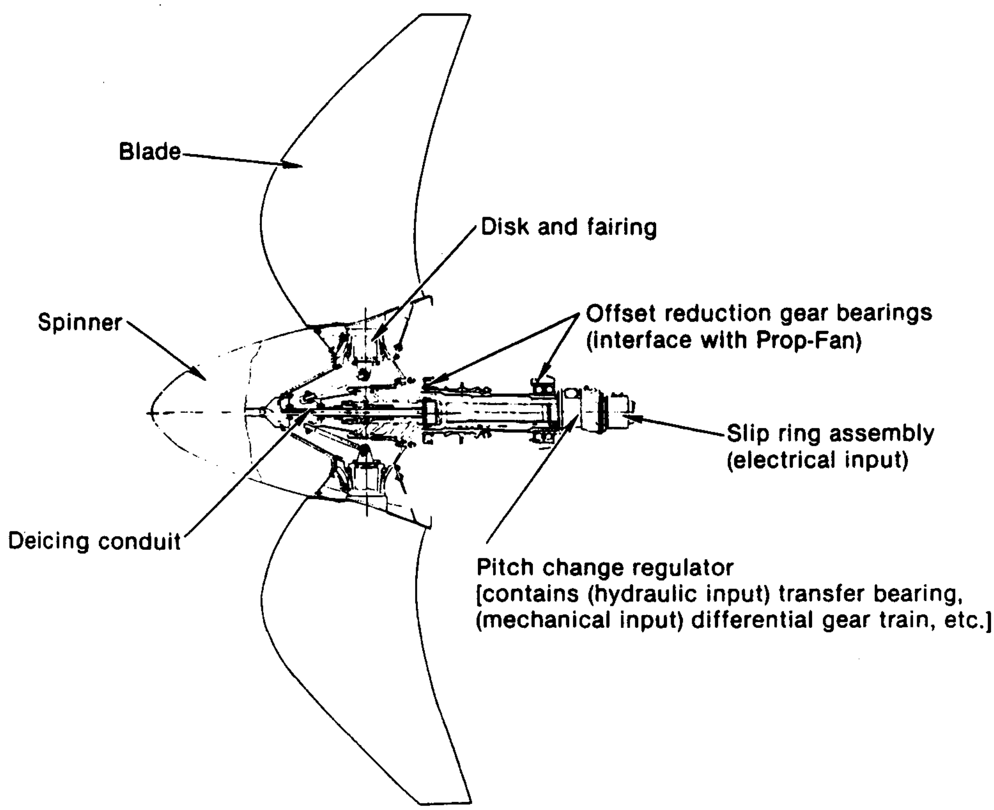


Figure 4.3-24 Prop-Fan Concept Description - This Prop-Fan can be used with an in-line or offset gearbox configuration. (J27638-18)

4.3.3 Integrated Propulsion System

In this section, conceptual nacelles, engine mounting concepts, acoustic treatment requirements, modular propulsion system concepts, and propulsion system reliability are discussed. Input from the aircraft manufacturers was incorporated in the evaluation of the integrated propulsion system.

4.3.3.1 Nacelle Conceptual Design

The nacelle designs were prepared to define mechanical aspects of the gearbox/nacelle interfaces. Figure 4.3-25 illustrates the conceptual nacelle design for an offset gearbox installation with the two-spool axial compression engine (STS678) and a chin (single) inlet. Figure 4.3-26 illustrates the conceptual nacelle for an in-line gearbox installation with the three-spool axial compression engine (STS679) and a bifurcated inlet. Both of these nacelles were included in the Propulsion System Integration Package. The choice of an over-the-wing installation reflects input from various airframe manufacturers.

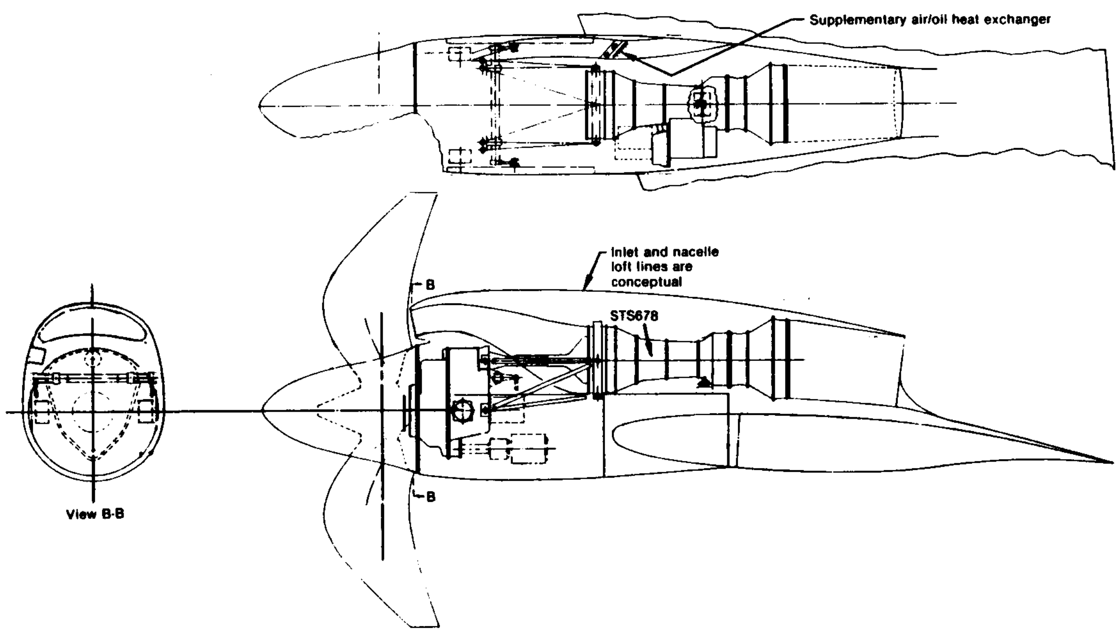


Figure 4.3-25 Conceptual Nacelle Design for an Offset Gearbox Installation - This design has a 0.32 diameter ratio due to the space required for the offset gearbox. (J27638-123)

Based on previous Hamilton Standard and NASA studies, a 0.24 spinner diameter (at the centerline of the blade root) to Prop-Fan blade diameter, set by Prop-Fan aerodynamic considerations, was used in the nacelle designs. The nacelle for the offset gearbox installation has been configured to have a 0.32 diameter ratio (maximum nacelle diameter to Prop-Fan blade diameter), based on the space available behind the Prop-Fan spinner for the offset compound idler gearbox. The nacelle for the in-line gearbox installation has been configured to have a 0.28 diameter ratio due to the smaller diameter of the split path in-line gearbox.

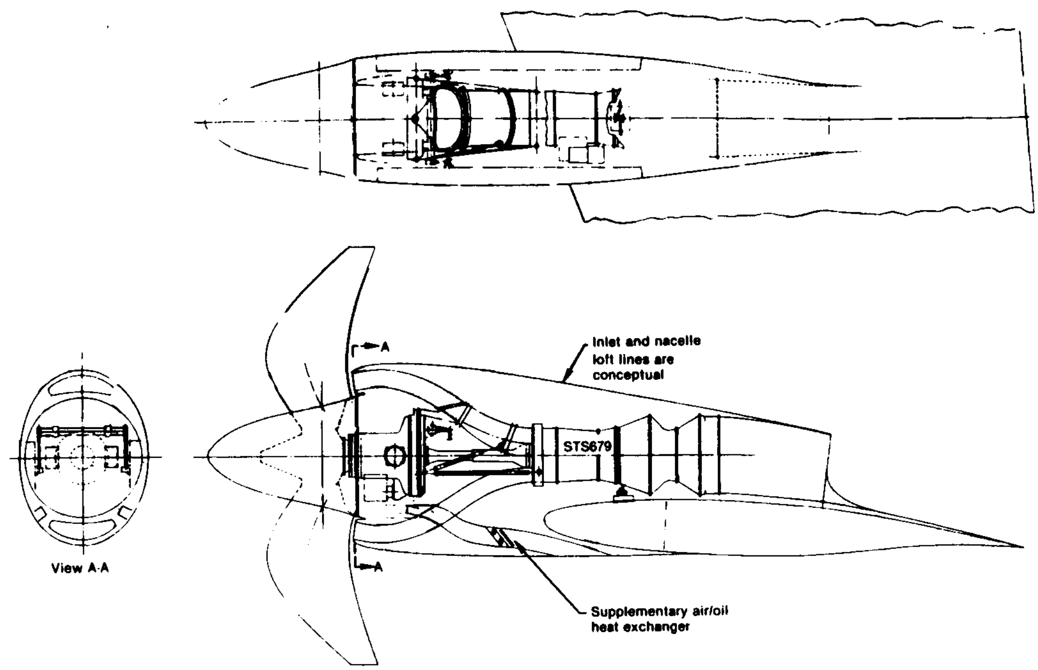


Figure 4.3-26 Conceptual Nacelle Design for an In-Line Gearbox Installation - This design has a 0.28 diameter ratio, due to the slimmer in-line gearbox. (J27638-122)

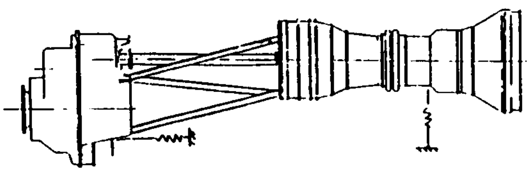
The external aerodynamic lines for the nacelle, which provide proper blockage for the Prop-Fan, are conceptual in nature. The final aerodynamic nacelle lines would be tailored to the flow field of the specific aircraft application. Detailed studies are being conducted by NASA and the airframe manufacturers to "tailor" the nacelle and aircraft wing to minimize aerodynamic interface losses. The nacelles identified in this study are intended to scope the mechanical design for use in the Engine/Aircraft Evaluation (Task IV).

It should be noted that the axial position of the Prop-Fan blade relative to the quarter chord of the wing shown in the figures was established by Hamilton Standard. The final axial location will be determined by integrated studies which consider propeller excitation, wing structure, maintenance, weight, and other aerodynamic/structural factors. These studies will require close coordination with airframe manufacturers. The axial location of the exhaust nozzle plane follows previous aircraft studies; thermal shielding may be required on the wing surface.

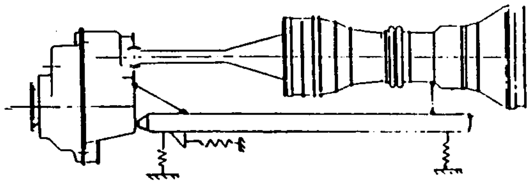
The over-the-wing "tractor" installation which was selected for both the in-line and offset gearbox nacelle concepts should provide adequate ground clearance for a typical low wing commercial aircraft. While a wing mount is typical of current propeller installations, there are other mount locations which may also be practical. Tail mounted engines, either pylon or horizontal stabilizer mounted, may offer significant cabin noise and/or aerodynamic benefits. An assessment of the nacelle/aircraft aerodynamic interactions leading to a minimum drag nacelle concept and the overall effects of engine location on airplane design and performance are beyond the scope of this study. These issues should be addressed in future studies involving both aircraft and engine manufacturers.

4.3.3.2 Propulsion System Mounting

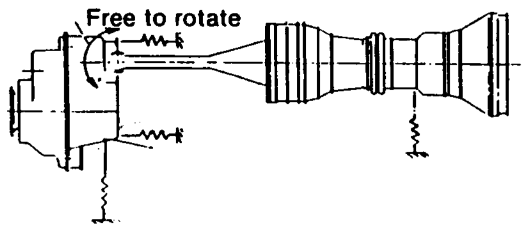
Three candidate propulsion system mounting schemes are shown in Figure 4.3-27. The "integrated" engine and reduction gear mount system ties the two units together structurally to form a single functional unit. With the "integrated nacelle," the gearbox and engine are mounted to a stiff frame which is in turn shock mounted to the airframe nacelle. The "partially independently" mounted gearbox and engine system requires stiff mounting of the gearbox and engine to the airframe to minimize deflections between the reduction gear and the engine. Based on a careful analysis of each configuration, and confirmation from the airframe manufacturers and the NASA Program Manager, the integrated engine/ reduction gear mount system was selected for the Propulsion System Integration Package. A discussion of the advantages and disadvantages of each system follows.



"Integrated" engine and reduction gearbox



"Integrated nacelle" (cradle mount)



"Partially independent" mounted gearbox and engine

Figure 4.3-27 Candidate Propulsion System Mounting Schemes - A variety of methods for mounting the propulsion system were evaluated. (J27638-19)

A schematic drawing of the "integrated" engine and reduction gear mounting system, highlighting the basic components, is presented in Figure 4.3-28. Two mount planes are provided -- one at the reduction gear and the other located at approximately the center of gravity of the engine. Additional weight will be required in the engine casing to minimize compressor tip clearance increases relative to an "integrated nacelle" configuration. The aircraft nacelle provides the primary mounting structure. The support structure consists of two box beams cantilevered forward of the wing box structure on either side of the powerplant joined together by a bulkhead attached to the reduction gear. The forward bulkhead provides a pick up point for the front mount plane while the structure attached to the wing box provides support for the rear mount plane.

A Prop-Fan torque reaction system is provided to handle the large Prop-Fan torque while allowing the mount to be sized for thrust, maneuver loads, and vibration isolation. The isolation of engine/Prop-Fan generated vibration is considered a major requirement for passenger comfort. The torque link system may be eliminated if vibration isolators can be made stiff enough to absorb Prop-Fan torque and can be oriented to permit the powerplant to translate freely in response to vibration while absorbing Prop-Fan torque.

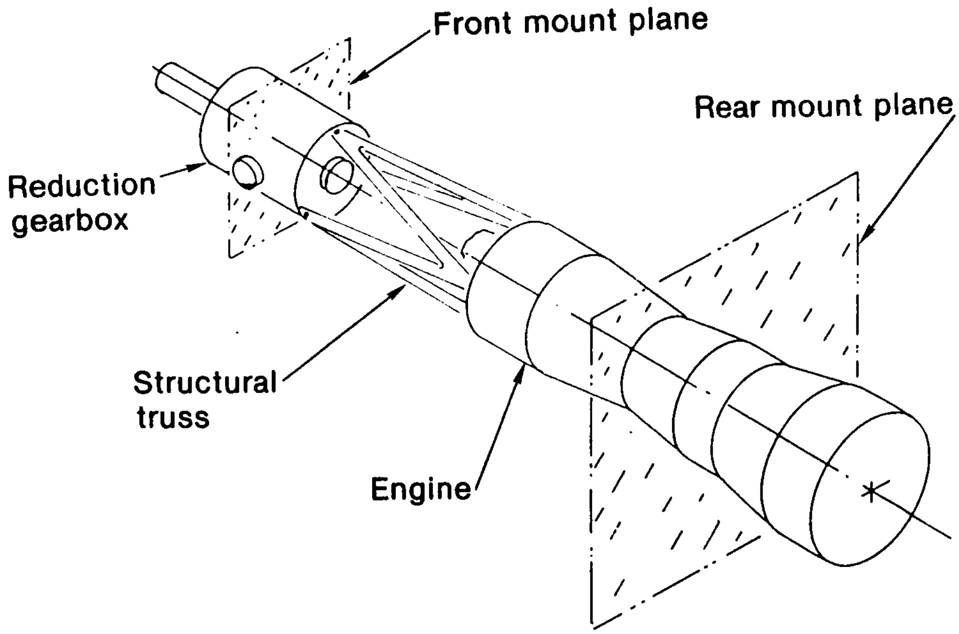


Figure 4.3-28 Schematic Drawing of the "Integrated" Engine and Reduction Gear Mounting System - This mounting scheme was the most promising candidate for a Prop-Fan propulsion system. (J27638-20)

The "integrated nacelle" mounting system is shown schematically in Figure 4.3-29. The gearbox and engine are mounted on a stiff frame which is shock mounted to the airframe nacelle. This system is designed to reduce structural case deflections, thus insuring optimum engine performance. Total propulsion system weight will be higher with this system than with the "integrated" mounting approach shown in Figure 4.3-28. As with the "integrated" engine and reduction gear mounting system, the airframe nacelle provides the primary mounting structure. The support structure consists of two box beams cantilevered forward of the wing box structure on either side of the powerplant joined by a bulkhead attached to the reduction gear. The forward bulkhead provides pick up points for the powerplant front mount plane while the structure attached to the wing box provides the rear mount plane. A Prop-Fan torque link system is provided to handle the Prop-Fan torque while allowing the shock mounts to be sized for thrust, maneuver reaction and vibration isolation.

A coordinated effort between engine and airframe manufacturers, beyond the scope of this contract, is required to avoid duplicate structures between the propulsion system and the aircraft.

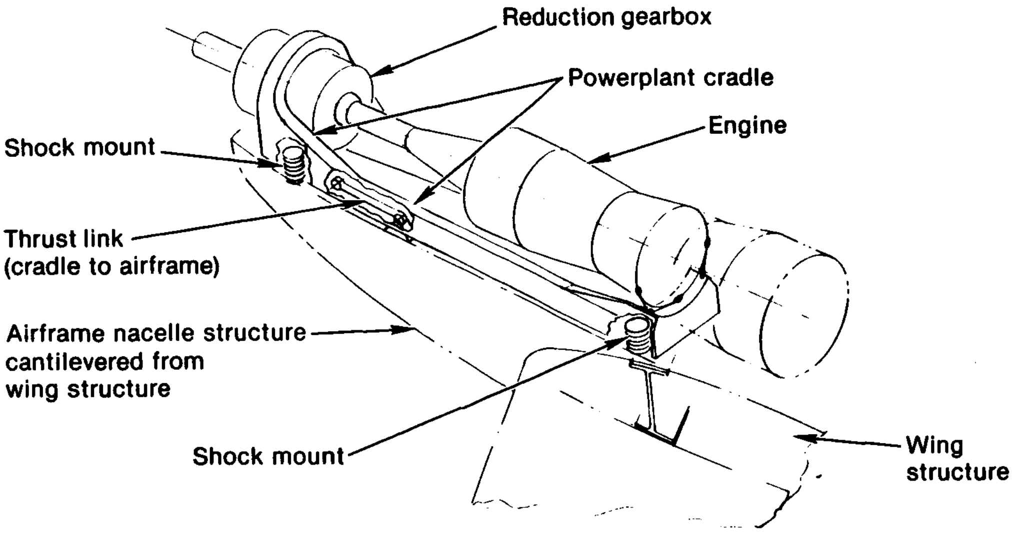


Figure 4.3-29 Schematic Drawing of "Integrated Nacelle" (Cradle) Mounting System - Propulsion system performance is improved with this configuration. (J27638-69)

A schematic drawing of the "partially independently mounted" reduction gear and engine mounting system is presented in Figure 4.3-30. As noted previously, the reduction gear must be stiff mounted to the airframe in order to minimize deflections between the gearbox and the engine. The stiffness required to minimize these deflections makes it difficult to provide vibration isolation. In addition, couplings are required between the engine and gearbox to handle the large deflections which result from the independent mounting of each component.

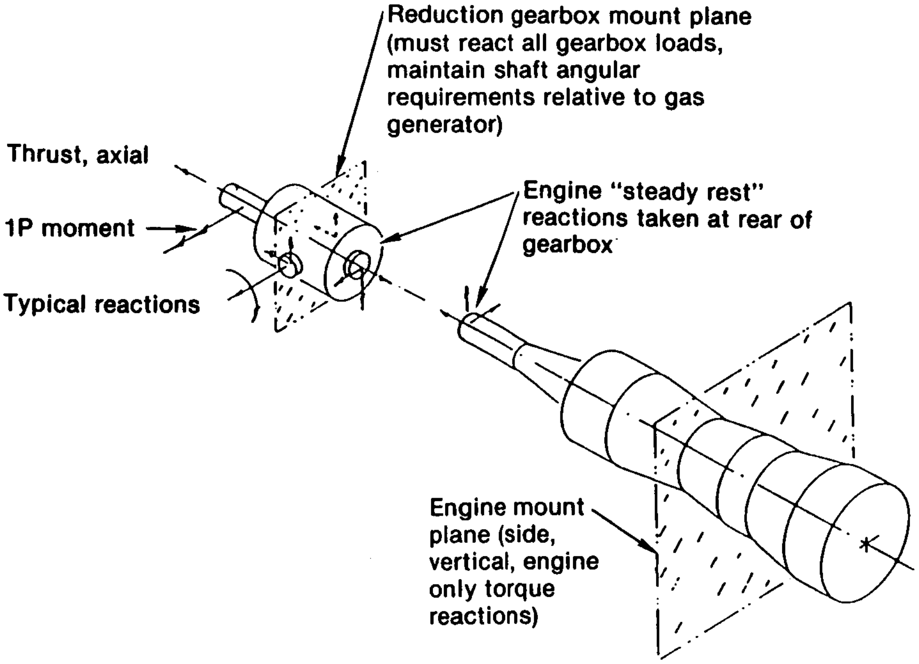


Figure 4.3-30 Schematic Drawing of the "Partially Independently Mounted" Reduction Gear and Engine Mounting System - This system was eliminated from consideration due to excessive vibration and the complexity of the design. (J27638-21)

Based on input from the aircraft manufacturers, the "integrated" engine and reduction gear mounting system was selected for the Propulsion System Integration Package. The mounting system for the offset gearbox configuration is illustrated in Figure 4.3-31. As shown in the figure, the offset gearbox and engine are tied together by a simple tubular truss system.

The "integrated" engine and reduction gear mounting system for the in-line gearbox configuration is shown in Figure 4.3-32. With this concept, a portion of the inlet duct is structurally tied to the engine and the gearbox to avoid structural links in the aerodynamic flowpath of the inlet. These links would result in small performance and engine inlet distortion penalties.

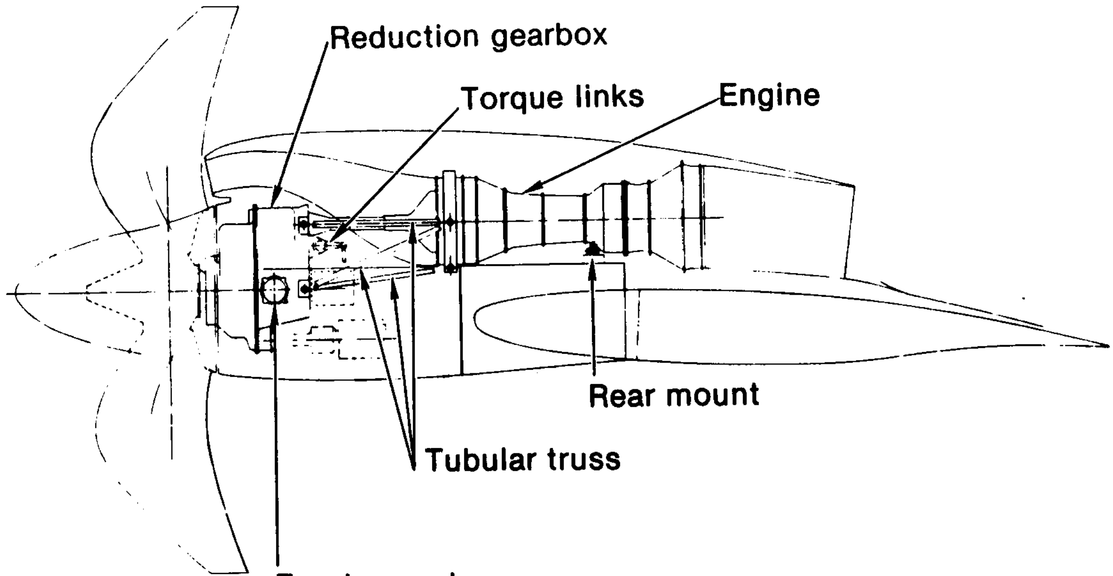


Figure 4.3-31 "Integrated" Engine and Reduction Gear Mounting System Selected for the Offset Configuration - This mounting system was included in the Propulsion System Integration Package. (J27638-13)

While the "integrated" engine and reduction gear mounting system has been selected for the Prop-Fan propulsion system, there are several technical issues which require study effort beyond the scope of the current contract. These issues are summarized in Table 4.3-XXV.

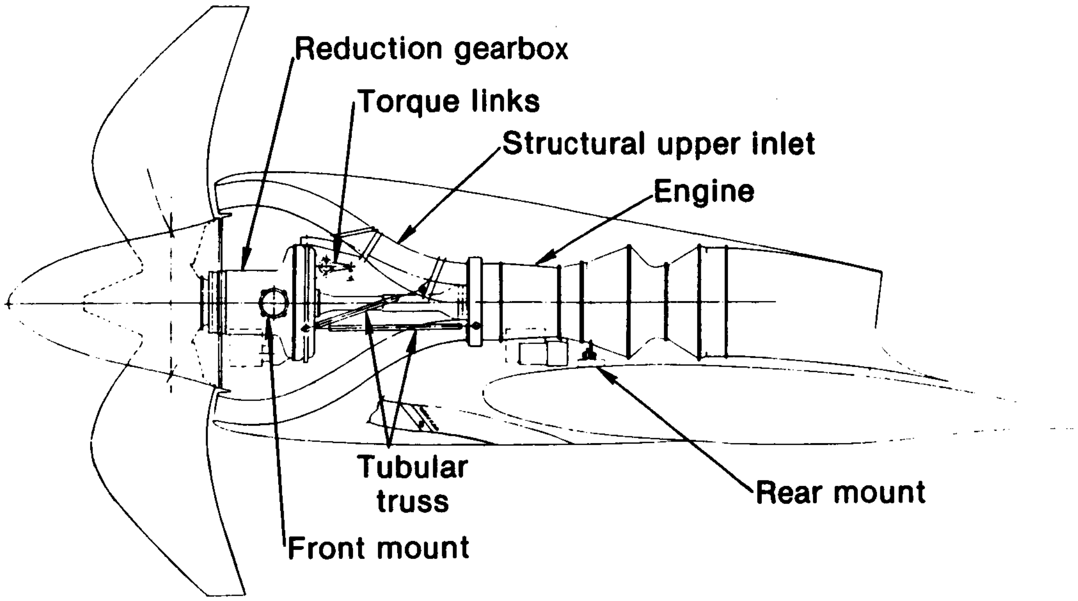


Figure 4.3-32 "Integrated" Engine and Reduction Gear Mounting System Selected for the In-Line Configuration - This mounting system was included in the Propulsion System Integration Package. (J27638-10)

TABLE 4.3-XXV
MOUNTING CONSIDERATIONS REQUIRING ADDITIONAL STUDY EFFORT

o Powerplant/aircraft structural dynamics studies

Axial location of engine relative to gearbox and wing box
Shock isolation trade studies
Effect of these factors on wing flutter

o Integrated engine and gearbox structure

Structural links between engine and gearbox
Primary structure with inlet between engine and gearbox

o "Integrated nacelle" (cradle mount) mounting engine/aircraft study

4.3.3.3 Acoustic Treatment Requirements

Studies performed by the Pratt & Whitney Acoustics Research Staff indicate that inlet and exhaust acoustic treatment will not be required to meet Federal Aviation Administration Stage 3 noise level requirements. The analysis performed during these studies considered the Prop-Fan, reduction gear and engine noise, including compressor, burner, and turbine noise generation. In Task IV, the engine configurations were evaluated to determine if any new noise sources could be identified which would require inlet, exhaust, or other acoustic treatment. No new sources requiring acoustic treatment were identified.

4.3.3.4 Modular Maintenance Concept

The Prop-Fan propulsion system employs a modular maintenance concept to maximize accessibility to propulsion system compounds, thus minimizing maintenance costs. Table 4.3-XXVI lists the major modules for an offset gearbox installation, an in-line gearbox installation, and the "non-free" and "free" power turbine engines. The elapsed times for removal and replacement of the major modules in the propulsion system are summarized in Table 4.3-XXVII. A more detailed description of the removal/replacement procedures follows.

TABLE 4.3-XXVI
MAJOR MODULES IN THE PROP-FAN PROPULSION SYSTEM

- Offset gearbox installation
  - Single prop-fan blade
  - Prop-fan pitch control regulator
  - Prop-fan module
  - Gearbox
  - Power shaft
  - Turboshaft engine
- In-line gearbox installation (modules uniquely different from offset gearbox installation)
  - Gearbox
  - Prop-fan pitch controls modules
  - Power shaft
- Turboshaft engine modules
  - "Non-free" power turbine (two-spool) engine — STS678
  - "Free" power turbine (three-spool) engine — STS679

Modules for an Offset Gearbox Installation

Figure 4.3-33 shows the six major modules in the Prop-Fan propulsion system with an offset gearbox: (1) single Prop-Fan blade, (2) Prop-Fan pitch control regulator, (3) Prop-Fan module, (4) gearbox, (5) power shaft, and (6) turboshaft engine. A graphic description of the removal procedure for each module follows. Modules which are common to both the offset and in-line installation are noted. Easy accessibility to the pitch change regulator makes the offset installation somewhat easier to maintain than the in-line installation.

TABLE 4.3-XXVII
TYPICAL PROPULSION SYSTEM COMPONENT
MAINTENANCE ACTION (REMOVAL/REPLACEMENT) TIMES

	Maintenance Action Elapsed Time, Minutes
o Offset Gearbox Installation	
Single Prop-Fan Blade	70
Prop-Fan Pitch Control Regulator	23
Prop-Fan Module	60
Gearbox	180
Power Shaft (Engine-Gearbox Connecting Shaft)	30
Turboshaft Engine	120
o In-line Gearbox Installation (Modules Uniquely Different From Offset Gearbox Installation)	
Gearbox	180
Prop-Fan Pitch Control Modules (Gearbox)	180
Power Shaft (Engine/Gearbox Connecting Shaft)	30

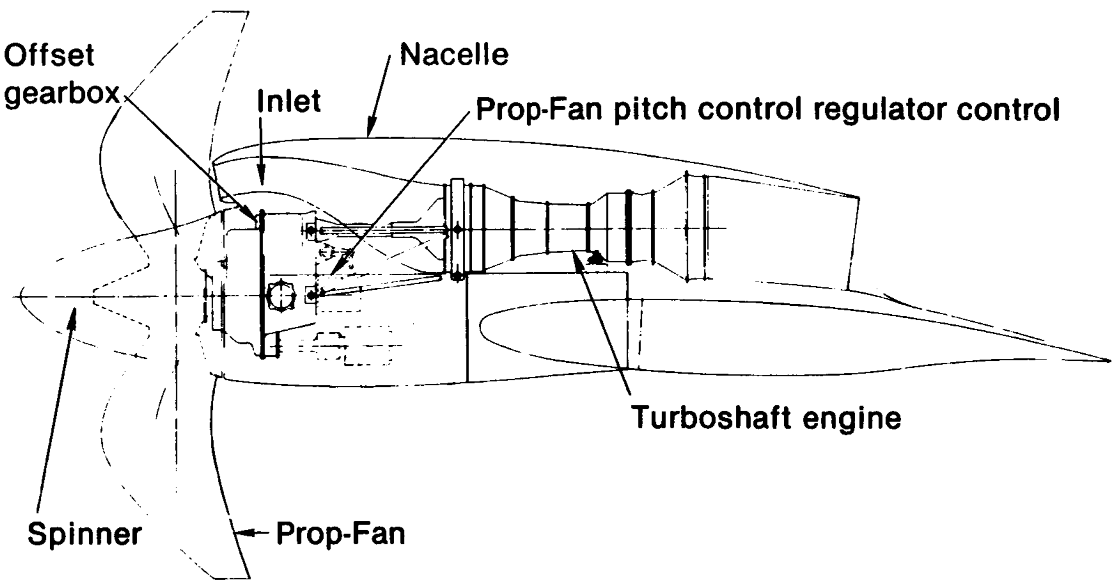


Figure 4.3-33 Propulsion System Components for the Offset Gearbox Installation -There are six major modules in the offset gearbox installation. (J27638-12)

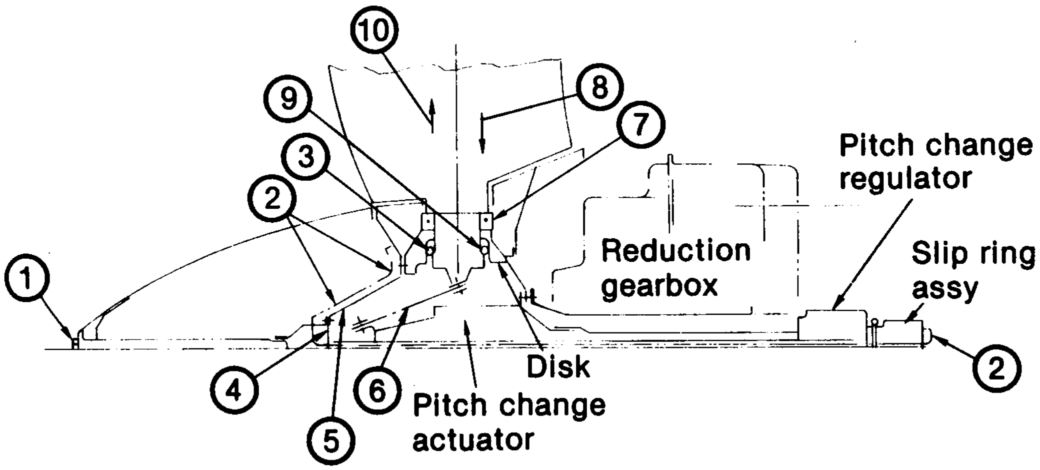
The sequence of events for the removal/replacement of a single Prop-Fan blade is shown in Figure 4.3-34. A typical elapsed time for removal/replacement is 70 minutes. This estimate includes the elapsed time to remove or replace the opposite moment weighted blade which is planned for this maintenance action.

The removal/replacement actions for the Prop-Fan blade pitch change regulator in the offset gearbox installation are shown in Figure 4.3-35. A typical elapsed time for removal/replacement is 23 minutes.

Figure 4.3-36 shows the procedures for removal or replacement of the Prop-Fan module. The same procedure is used for either an in-line or offset gearbox installation. Typical elapsed time to remove/replace the Prop-Fan module is 60 minutes.

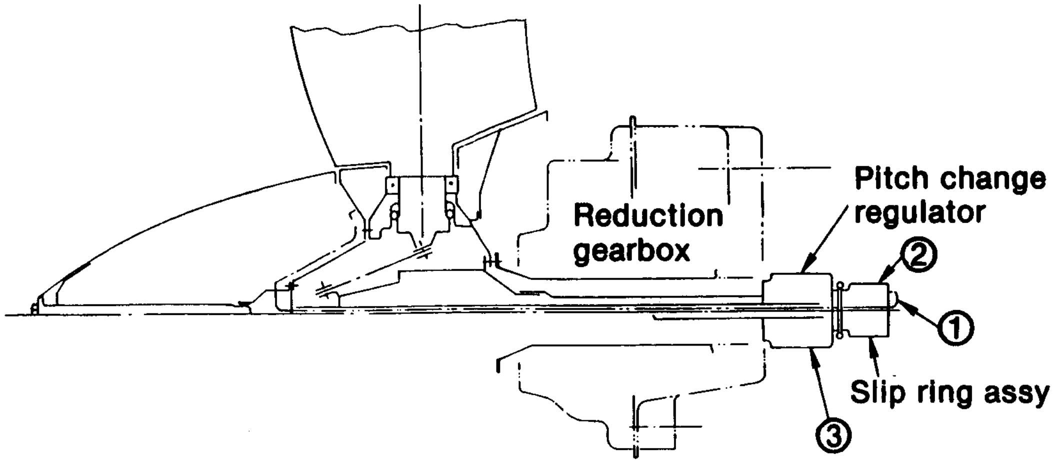
The removal/replacement procedures for the offset gearbox are shown in Figure 4.3-37. A typical elapsed time required to remove or replace the reduction gear is 180 minutes.

The removal of the engine/gearbox connecting shaft for either the offset or in-line reduction gear system is shown in Figure 4.3-38. Typical elapsed time is 30 minutes.



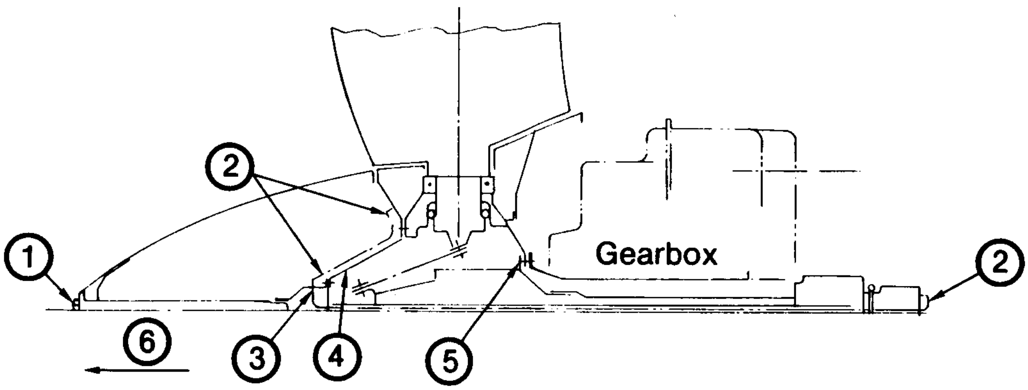
1 Remove spinner bolt and spinner
2 Disconnect deicer leads
3 Drain bearing oil
4 Remove deicer conduit assy
5 Remove disk cover
6 Detach link
7 Remove outer blade clamp
8 Slide blade in toward centerline
9 Remove blade retention ball assy
10 Withdraw blade

Figure 4.3-34 Single Prop-Fan Blade Removal - The average time for removal and replacement is 70 minutes. (J27638-22)



(1) Disconnect deicer leads
(2) Remove slip ring assy
(3) Remove pitch change regulator

Figure 4.3-35 Pitch Change Regulator Removal; Offset Gearbox Installation - The average time for removal and replacement is 23 minutes. (J27638-23)



(1) Remove spinner bolt and spinner
(2) Disconnect deicer leads
(3) Remove deicer conduit assembly
(4) Remove disk cover
(5) Remove bolts from gearbox output shaft
(6) Remove Prop-Fan assembly

Figure 4.3-36 Prop-Fan Module Removal - The average time for removal or replacement is 60 minutes. (J27638-25)

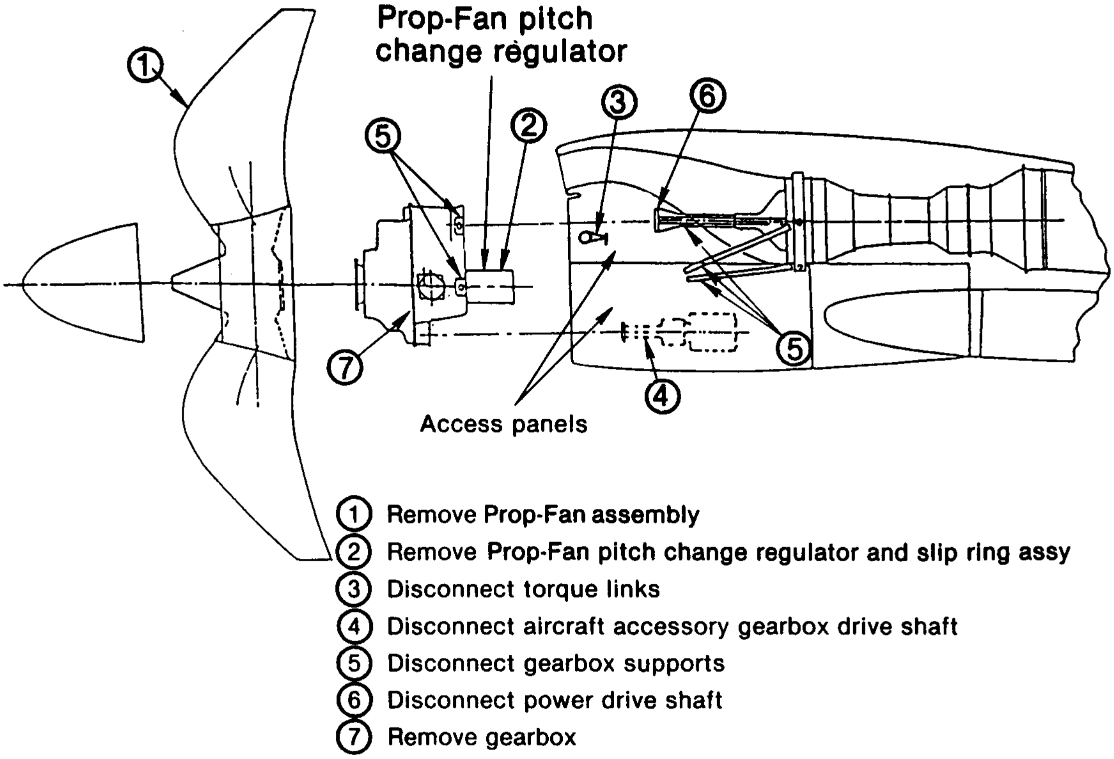


Figure 4.3-37 Offset Gearbox Removal - The average time for removal and replacement is 180 minutes. (J27638-26)

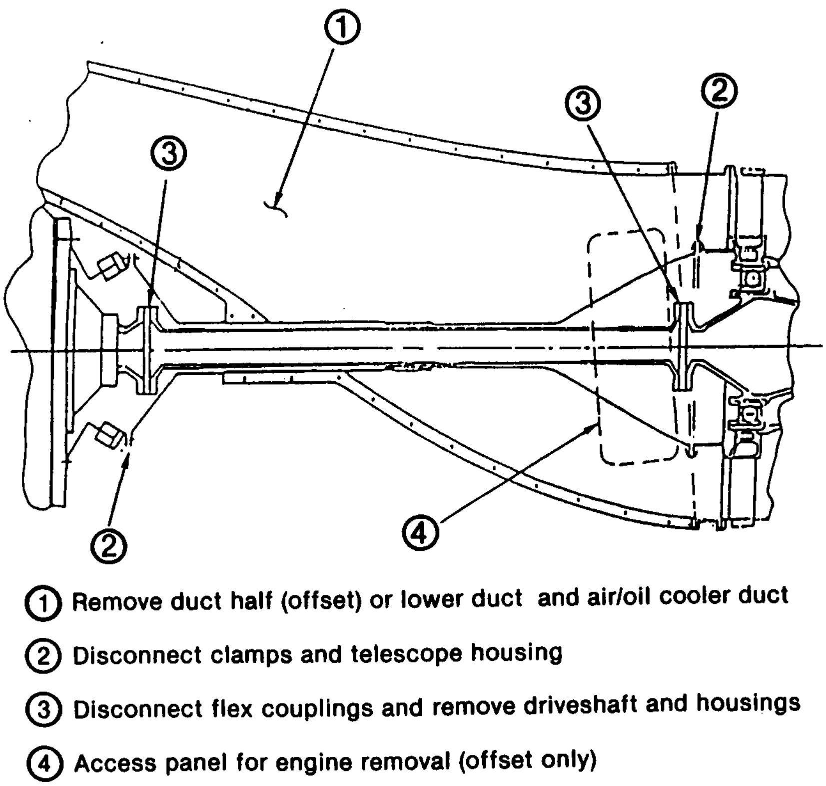


Figure 4.3-38 Power Shaft Removal - The average time for removal and replacement is 30 minutes. (J27638-28)

Turboshaft engine removal for an over-the-wing installation with either the in-line or offset gearbox is shown in Figure 4.3-39. Typical elapsed time for removal/replacement of the turboshaft engine is 120 minutes.

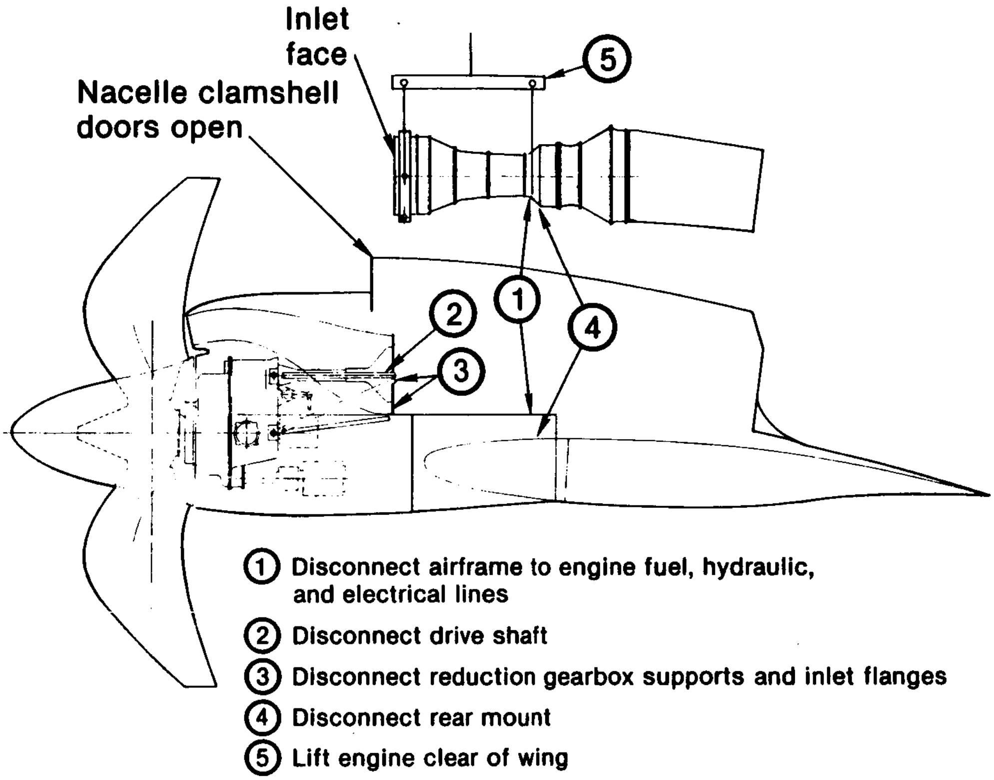


Figure 4.3-39 Turboshaft Engine Removal - The average time for removal and replacement is 120 minutes. (J27638-29)

In-Line Propulsion System Modules

The major modules in a Prop-Fan propulsion system with an in-line gearbox are shown in Figure 4.3-40. Most of the modules are similar to those in the offset gearbox installation. However, there are significant differences in the gearbox/pitch change regulator. In the in-line gearbox system, the pitch change regulator is divided into two modules. One of the modules contains control hardware such as an electro hydraulic valve, linear variable differential transfer and a hydraulic motor. This module is mounted on the gearbox and can be removed without removing the entire gearbox. The other module is located inside the gearbox and contains a differential geartrain and hydraulic transfer bearing. The gearbox must be removed to gain access to this module. This lack of accessibility to the pitch change control has an adverse effect on maintenance cost. It is recommended that additional studies be conducted to increase accessibility to the pitch control.

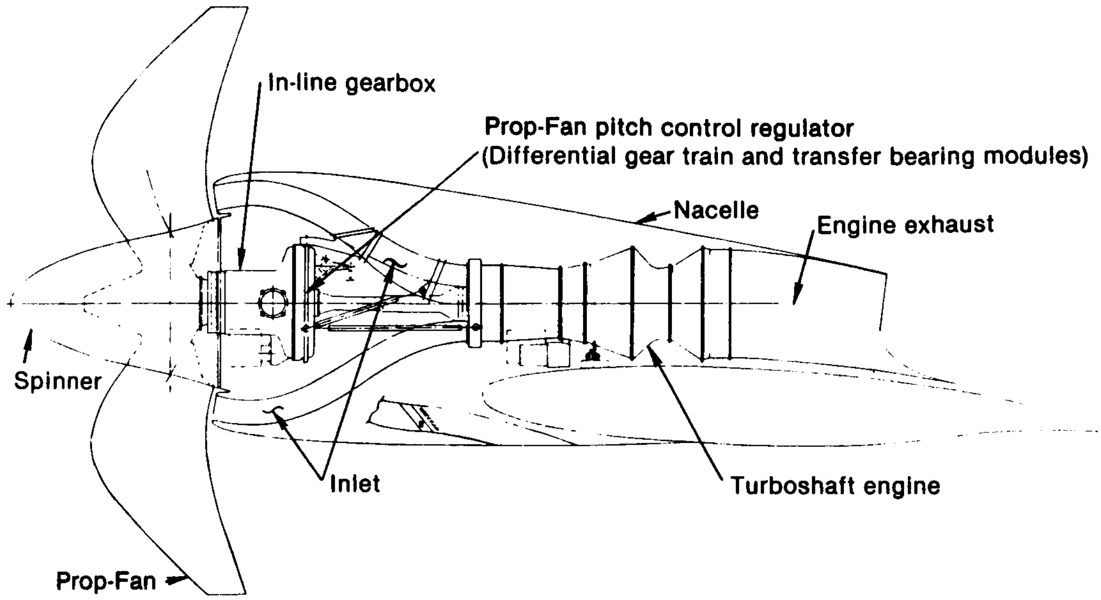


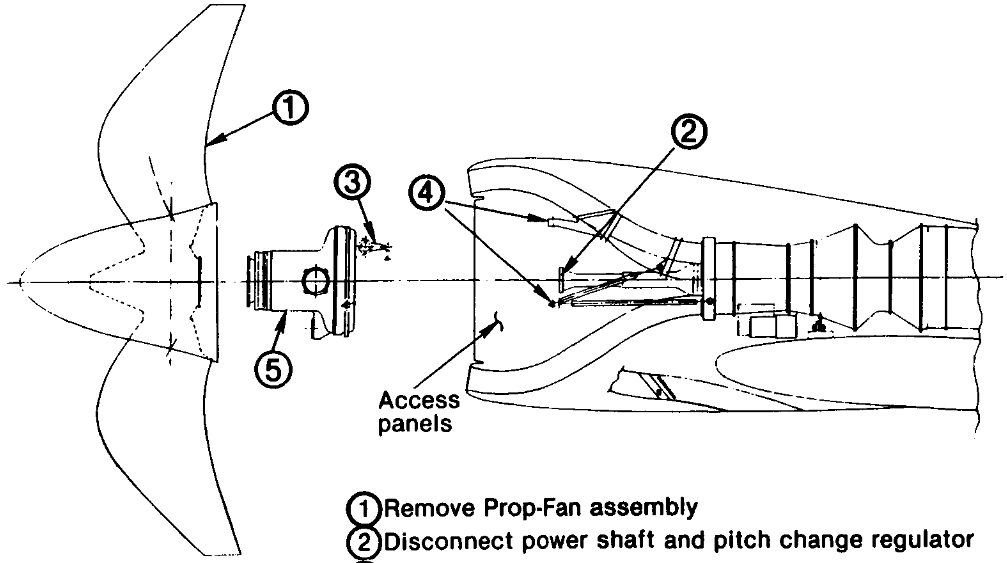
Figure 4.3-40 Major Propulsion System Modules; In-Line Gearbox Installation - The modules in the in-line installation are similar to those in the offset installation except for the gearbox and pitch change regulator (J27638-9)

The removal procedure for the in-line gearbox is shown in Figure 4.3-41. The removal procedure for the pitch change regulator is shown in Figure 4.3-42. The gearbox must be removed in order to remove the Prop-Fan pitch change regulator. However, while the procedures are somewhat different from the offset gearbox, the typical elapsed time for removal and replacement is the same - 180 minutes.

Turboshaft Engine Modules

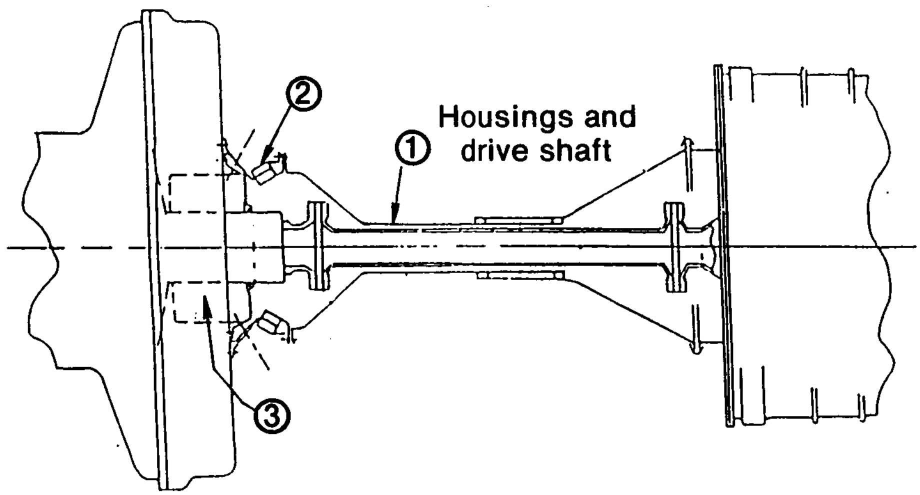
A modular concept was also devised for the turboshaft engines in order to maximize accessibility and minimize maintenance costs. One of the primary objectives of engine modularity is to provide quick access to the hot section, including the combustor and high-pressure turbine. This access is provided through simple removal procedures for the power and low turbine modules.

The modular maintenance concept for the "non-free" power turbine (two-spool) engine (STS678) is illustrated in Figure 4.3-43. The engine system has six modules including the inlet case and low compressor, intermediate case, high compressor, combustor and high-pressure turbine vanes, high-pressure turbine, power turbine and shaft. Additional components not shown on the figure which are considered modules include the engine accessory gearbox and the propulsion system electronic control.



①Remove Prop-Fan assembly
②Disconnect power shaft and pitch change regulator
③Disconnect torque links
④Disconnect gearbox supports
⑤Remove gearbox

Figure 4.3-41 In-Line Gearbox Removal - The average time for removal and replacement is 180 minutes. (J27638-27)



① Remove housings and drive shaft
② Remove gimbal assy and rear outer housing
③ Remove pitch change regulator and input drive pinion

Figure 4.3-42 Pitch Change Regulator Removal - Since the gearbox must be removed, the average time to remove and replace the pitch change regulator is also 180 minutes. (J27638-24)

Several major engine subassemblies can be removed for inspection, repair, or replacement. Among the external accessories in this category are the angle gearbox and shaft, ignitor box and ignitor plug, the anti-icing shutoff valve, and the ignition harness. Removable lubrication system subassemblies include the oil tank, oil filters, oil pressure and flow transmitters, and the oil quantity transmitter. Modular components in the propulsion system control include the electronic engine control and associated dedicated generator, the fuel pump, fuel control and distribution valves, the fuel heater and valve, fuel filter and fuel flow transmitter, several temperature and pressure sensors, and the rotor speed tach generators.

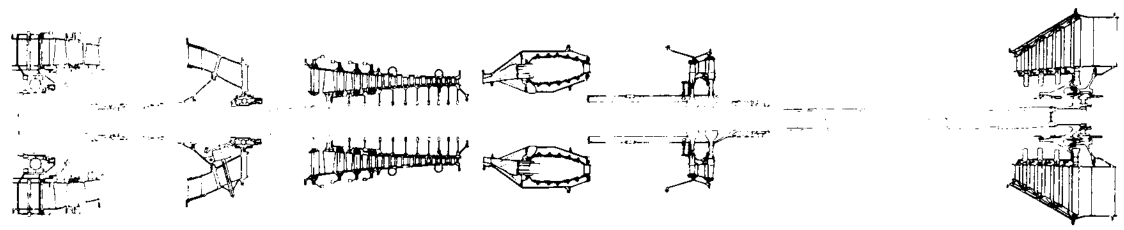


Figure 4.3-43 Turboshaft Engine Modules; "Non-Free" Power Turbine (Two-Spool) Engine - The STS678 engine system is comprised of six major modules. (J27638-70)

The modular maintenance concept for the "free" power turbine (three-spool) engine (STS679) is illustrated in Figure 4.3-44. The engine has seven modules including the inlet case and low compressor, intermediate case, high compressor, combustor and high pressure turbine vanes, high pressure turbine, intermediate turbine and shaft, and power turbine and shaft. The power turbine module includes the shaft and the forward thrust bearing. As with the STS678 engine, the engine accessory gearbox and the propulsion system electronic control are separate modules, which are not shown in the figure.

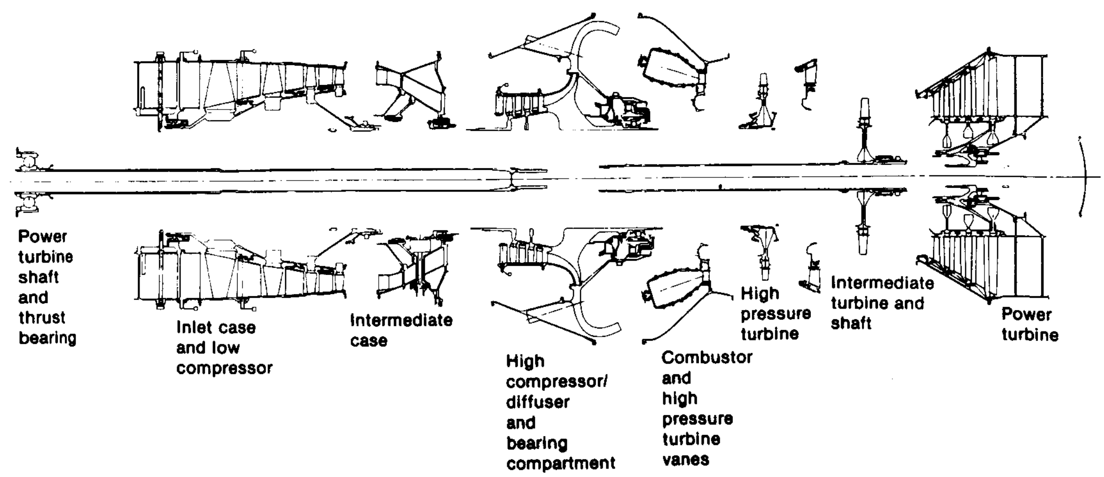


Figure 4.3-44 Turboshaft Engine Modules; "Free" Power Turbine (Three-Spool) Engine - The STS679 engine system is comprised of seven major modules. (J27638-71)

4.3.3.5 Propulsion System Reliability

The reliability of the integrated Prop-Fan propulsion system is determined by evaluating three key components: the Prop-Fan, reduction gear system, and the turboprop engine. Table 4.3-XXVIII provides reliability estimates for the individual components and the total Prop-Fan propulsion system. Information on an integrated propulsion system for the reference turbofan engine is also included in the table. The estimates are based on mean time between removals (MTBR), a statistical measure of reliability used to develop maintenance cost data. As indicated in the table, the reliability of the two systems is comparable.

TABLE 4.3-XXVIII
PROPULSION SYSTEM RELIABILITY PREDICTION

	Mean Time Between Removal, Hours	
	Chargeable	All Causes
Major Prop-Fan Propulsion System Modules		
o Prop-Fan	18,400	2700
o Reduction Gear		
- In-Line	15,000	7000
- Offset	32,000	9400
o Turboshaft Engine		
- STS678 (Two-Spool)	6900	630
- STS679 (Three-Spool)	6550	620
Integrated Prop-Fan Propulsion System		
o STS678 With In-Line Gearbox	3750	480
o STS678 With Offset Gearbox	4350	490
Major Turbofan Propulsion System Modules		
o Reference Turbofan Engine, STF686		
- Without Reverser	6900	630
- Reverser Alone	8000	1900
Integrated Turbofan Propulsion System	3700	475

The reliability prediction for the turbofan engine (6900 hours not including the reverser and 3700 hours including the reverser) is compatible with current turbofan experience.

The reliability prediction for the integrated Prop-Fan propulsion system (including the Prop-Fan as a reverser) is essentially equal to the reliability prediction for the turbofan.

The mean time between removals is calculated by summing the projected failure rates of key parts and taking the reciprocal. The mature reliability level is predicted because it reflects the basic capability of the design and is the level which prevails during most of the useful life of the system. The prediction does not imply the existence of an exponential failure rate, but instead uses contributing factors such as mixture of parts prior to wear out, mix of old and new parts, and the inherent randomness of failures occurring in the tail ends of the standard statistical distribution. The system reliability prediction is based on a review of previous designs for which experience has been accumulated in both commercial and military applications. Once the basic component history has been selected, adjustment factors, based on engineering judgement, are developed to account for anticipated differences in design parameters such as speed, flow, and maintenance philosophy. These adjustment factors are applied to the basic component failure rates to obtain a projected system failure rate.

There are two categories of MTBR estimates in Table 4.3-XXVIII; Chargeable and All Causes. Chargeable or "basic" removals are unscheduled removals of major modules which can be charged to the hardware. Causes of chargeable removals include manufacturing errors, quality control problems, or design flaws. Removals for all causes include chargeable events plus nonchargeable events: unscheduled removals of flight-line replaceable modules which can not be charged to the hardware. Examples include foreign object damage, maintenance damage, and unsubstantiated removals. Maintenance cost estimates cover both chargeable and nonchargeable events.

A somewhat more detailed discussion of the reliability estimates for the three key components in the Prop-Fan propulsion system follows.

Prop-Fan Reliability

The predicted mean time between chargeable removals for the ten-bladed Prop-Fan is 18,400 hours. A chargeable maintenance event involves removal of the Prop-Fan module, including the hub, pitch change mechanism, and Prop-Fan blades. The projected mean time between removals for all causes is 2697 hours. This category includes both chargeable and nonchargeable removals. Examples of the latter include removal or replacement of the spinner or pitch change regulator; maintenance actions which can be accomplished without removing the Prop-Fan module from the aircraft.

The reliability estimates for the Prop-Fan module are based on an assessment of the reliability of the individual parts, as defined on preliminary design layouts. The data reflect Hamilton Standard experience with comparable components.

Reduction Gearbox Reliability

A detailed evaluation of reduction gearbox reliability is presented in Table 4.3-XXIX. The mean time between removal for the primary transmission unit was determined separately by technical experts from Pratt & Whitney and Sikorsky. The answers were then compared. The independent MTBR prediction made by each technical team was essentially the same.

The impacts of aircraft accessory pads, Prop-Fan pitch control accessibility, and opposite hand rotation gearbox design are highlighted in the table.

TABLE 4.3-XXIX
REDUCTION GEAR RELIABILITY (MTBR)

	Reduction Gear Type	
	In-line Split Path	Offset Compound Idler
o Primary Transmission Unit, Mean Time Between Removal (MTBR), Hours	23,200*	42,000*
o Impact of One Aircraft Accessory Pad - MTBR, Hours	20,800	35,000
o Impact of Propeller Pitch Control Accessibility - MTBR, Hours	15,000	32,700
o Impact of Opposite Rotation - MTBR, Hours	15,000	27,300

* Includes projected failure rates of bearings, gears and seals.

Turboshaft Engine Reliability

Individual lives of critical parts in the two-spool and three-spool engines, including bearings, seals, blades, vanes, rotors, and structural support cases, were assessed and compared. The reliability of the "non-free" power turbine (two-spool) engine (STS678) was found to the about 5% greater than the reliability of the "free" power turbine (three-spool) engine (STS679). The chargeable mean time between removals for the two engines is 6900 and 6550 hours, respectively.

4.3.4 Propulsion System Integration Package

After the technical effort described in Sections 4.3.1 and 4.3.2 had been completed, a Propulsion System Integrated Package was prepared. The integration package permits comprehensive analysis of the characteristics of the Prop-Fan propulsion system, as well as comparative evaluation with a turbofan propulsion system incorporating comparable technology. The package covers the following major items: two-spool and three-spool engine configurations, gearboxes, aircraft accessory locations for power extraction, inlet configurations, a typical oil cooler arrangement, a Prop-Fan/engine control concept, propeller (Prop-Fan and pitch control configurations), conceptual nacelle, a typical engine mounting arrangement, and modular maintenance concept.

More specifically, the Propulsion System Integration Package includes: (1) a conceptual design drawing of the two turboprop propulsion system concepts, and (2) a base size turboprop engine propulsion system data pack and computer deck (including user manual for performance, dimensional, weight, and scaling information) and (3) a reference turbofan propulsion system data pack and computer deck. Acquisition cost and maintenance cost data for the turboprop and reference turbofan engines are provided in a separate document.

This section contains a brief description of the individual components in the Propulsion System Integration Package.

4.3.4.1 Conceptual Design Drawings of Prop-Fan Propulsion System

Separate drawings were prepared for the Prop-Fan propulsion system with an in-line gearbox and the Prop-Fan propulsion system with an offset gearbox. A conceptual design drawing of the propulsion system installation with an in-line gearbox is presented in Figure 4.3-45. The conceptual drawing illustrates the installation for both the "free" power turbine engine (STS679) and the "non-free" power turbine engine (STS678). The figure summarizes technical information developed during Task III and covers the conceptual nacelle (0.28 maximum nacelle diameter to Prop-Fan blade diameter), mounting, two possible locations for airframe accessories, bifurcated inlet, and the supplementary air/oil heat exchanger required for the oil cooler system.

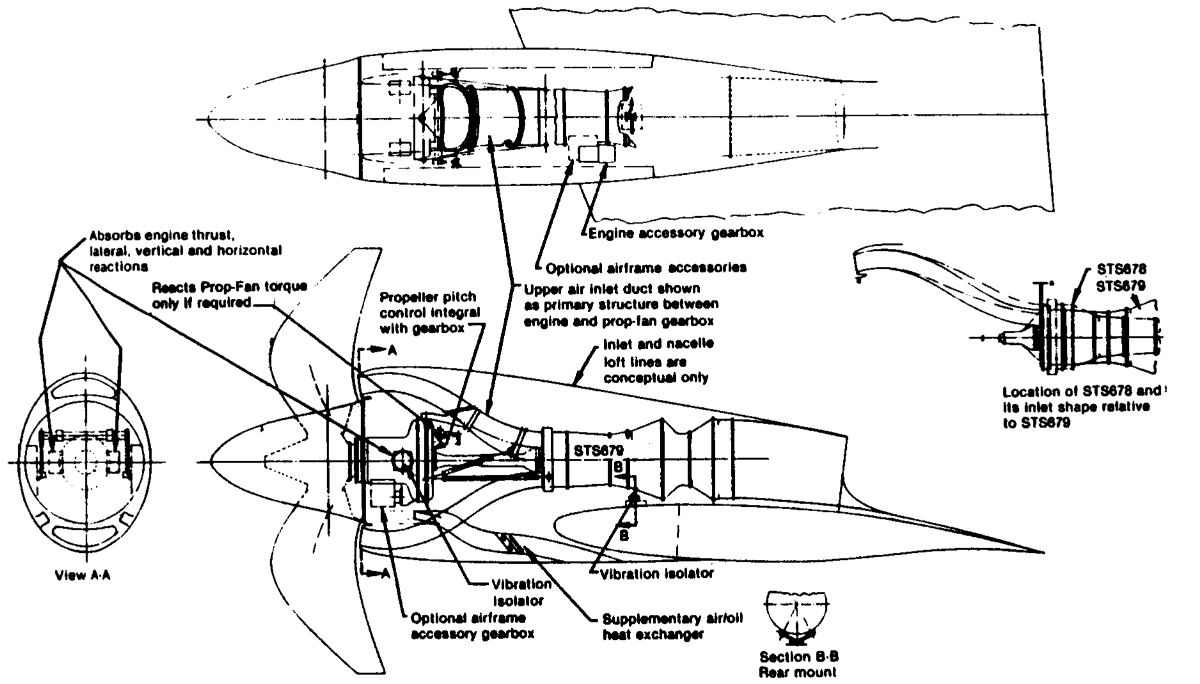


Figure 4.3-45 Conceptual Design Drawing of the Prop-Fan Propulsion System; In-Line Gearbox Installation - The drawing covers both the "free" and "non-free" power turbine engine configurations. (J27638-85)

Figure 4.3-46 shows the conceptual design of the Prop-Fan propulsion system installation with an offset gearbox. It highlights both the STS678 and STS679 engine configurations, a chin (single) inlet, nacelle (0.32 maximum nacelle diameter to Prop-Fan blade diameter), mounting, possible locations for aircraft accessories, chin inlet, and the supplementary air/oil heat exchanger required for the oil cooler system.

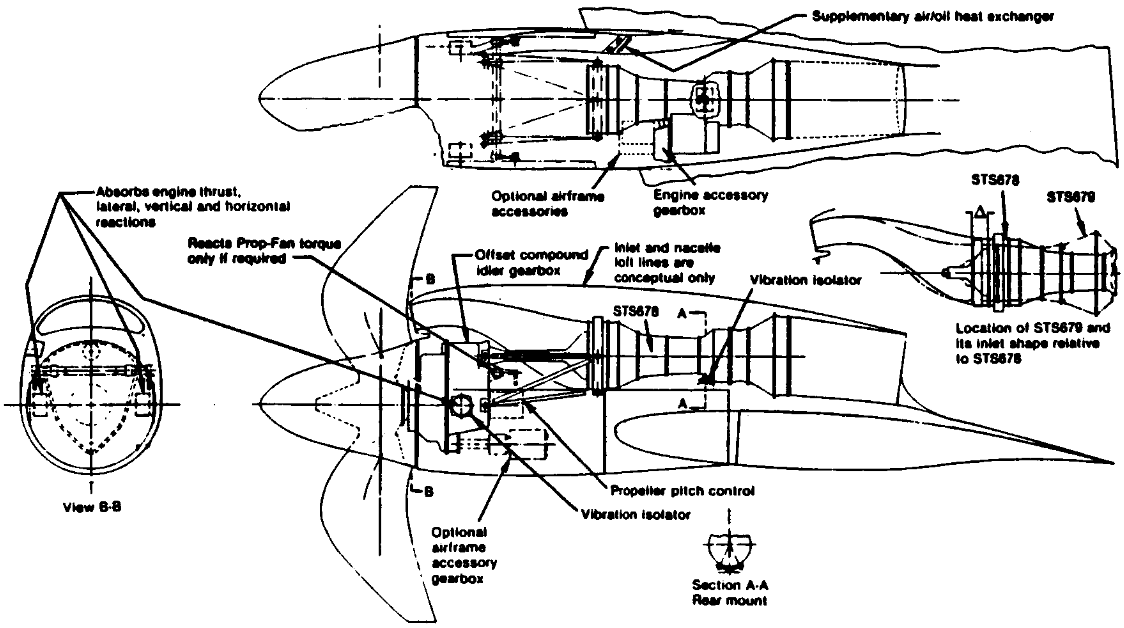


Figure 4.3-46 Conceptual Drawing of the Prop-Fan Propulsion System; Offset Gearbox Installation - The drawing covers both the "free" and "non-free" power turbine engine configurations. (J27638-84)

4.3.4.2 Base Size Turboprop and Reference Turbofan Engine Comparison

Table 4.3-XXX compares critical parameters for the turboprop engine and the reference turbofan. The reference turbofan has a higher pressure ratio than the turboprop engine which is consistent with the differences in engine core size (indicated by the high compressor exit corrected airflow). The high spool of the STF686 reference turbofan is essentially a scaled STS678 turboprop engine high spool.

The typical cruise part power performance for the turboprop and turbofan engines is shown in Figure 4.3-47. An indicated in the figure, the turboprop engine has a 16.6% advantage in thrust specific fuel consumption over the turbofan at the maximum cruise rating.

TABLE 4.3-XXX
BASE SIZE TURBOPROP AND REFERENCE TURBOFAN COMPARISON

	Turboprop Engine STS678	Reference Turbofan STS686
Takeoff Thrust @ M = .22 STD +14°C (+25°F) Day, N (lb)	69,836 (15,700)	68,146 (15,320)
Horsepower	11,800	
Maximum Climb Thrust @ 10668 m (35,000 ft), M = .75 STD	1824 (4020)	2155 (4750)
Overall Pressure Ratio @ Design Point (10,668 m (35,000 ft), M = .75 STD, 90% of Max Cruise)	34	37
Maximum Takeoff Combustor Exit Temperature (Sea Level, STD +14°C (+25°F) Day),		
Growth Rating - °C (°F)	1427 (2600)	1460 (2660)
Initial Rating - °C (°F)	1388 (2530)	1421 (2590)
Maximum Climb Combustor Exit Temperature STD - °C (°F)	1318 (2405)	1243 (2270)
Maximum Cruise Combustor Exit Temperature STD - °C (°F)	1274 (2326)	1206 (2203)
Pressure Ratio Split - Low Compressor plus Fan ID/High Compressor	2.0/17	2.2/17
Core Size, High Compressor Exit Corrected Airflow, kg/sec (lb/sec)	1.39 (3.06)	1.97 (4.4)
Fan Pressure Ratio	"1.06"	1.66
Jet Velocity Ratio	0.7	0.7
Bypass Ratio	"90"	7.0
Fan/Propeller Diameter - m (ft)	4.05 (13.3)	2.20 (4.86)

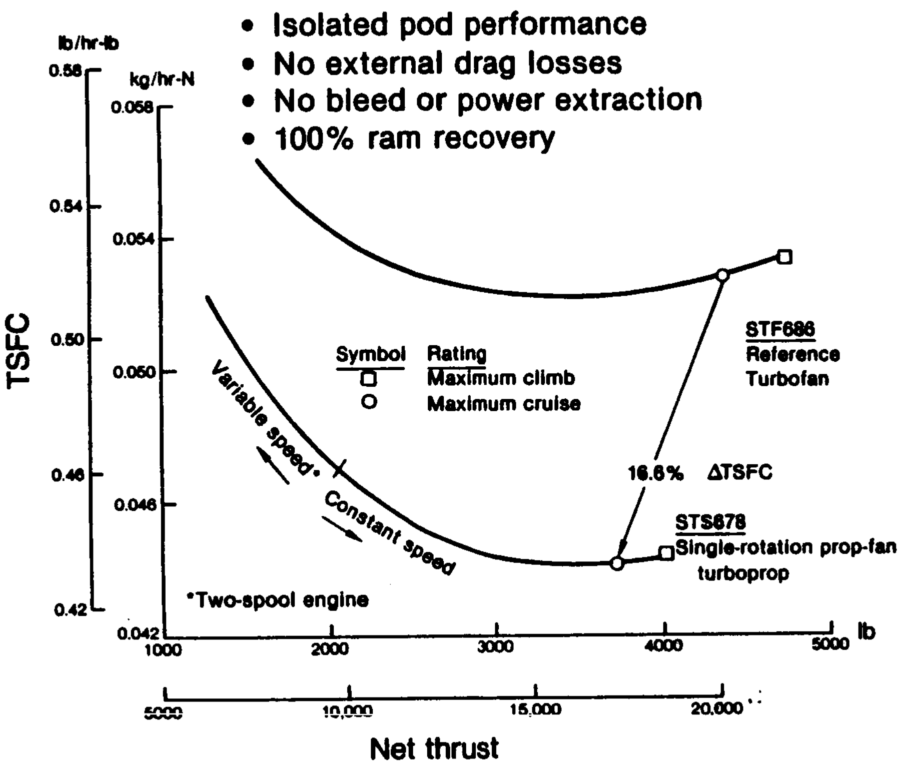


Figure 4.3-47 Part Power Performance Comparison - The turboprop has a 16% TSFC advantage over the turbofan. (J27638-203)

4.3.4.3 Turboprop Engine Propulsion System Data Package and Computer Deck (User Manual)

The capabilities and options included in the computer deck for the turboprop engine (STS678/STS679) are summarized in Table 4.3-XXXI. The more significant capabilities include: (1) the ability for the user to perform Prop-Fan tip speed and loading trade studies, (2) the ability to compare propulsion systems with single or counter rotation Prop-Fans, and (3) the ability to perform trade studies between constant and variable speed Prop-Fan operation. The user manual provides weight, dimensions, and scaling information for the various engine and gearbox options.

Base size dimensions for the "non-free" power turbine (two-spool) engine (STS678) with all axial compression and the "free" power turbine (three-spool) engine (STS679) with axial/centrifugal compression are presented in Figure 4.3-48.

Scaling curves for turboprop engine weight, dimensions, and performance are presented in Figure 4.3-49.

TABLE 4.3-XXXI
TURBOPROP ENGINE (STS678/679) COMPUTER DECK CAPABILITIES AND OPTIONS

- o Two-spool (STS678) or three-spool (STS679) engine approximation.
- o Ability for user to perform propeller tip speed and loading trade studies.
- o Constant speed/variable speed Prop-Fan option.
- o Option for user to specify temperature and pressure rise across Prop-Fan(s).
- o Single rotation and counter rotation Prop-Fan(s) options.
- o Gearbox efficiency input option.
- o Ability to alter propeller efficiency.
- o Capability for user to specify inlet and exhaust systems.
- o Standard customer bleed and power extraction options.
- o Capability to schedule environmental control system bleeds with flight conditions.
- o Capability to operate as a stand-alone program or as a subroutine to the user's program.
- o Capability to automatically sequence the input to allow rapid generation of performance information at a series of flight conditions.
- o User Manual provides weights, geometric dimensions and scaling curves for engine and gearbox options.

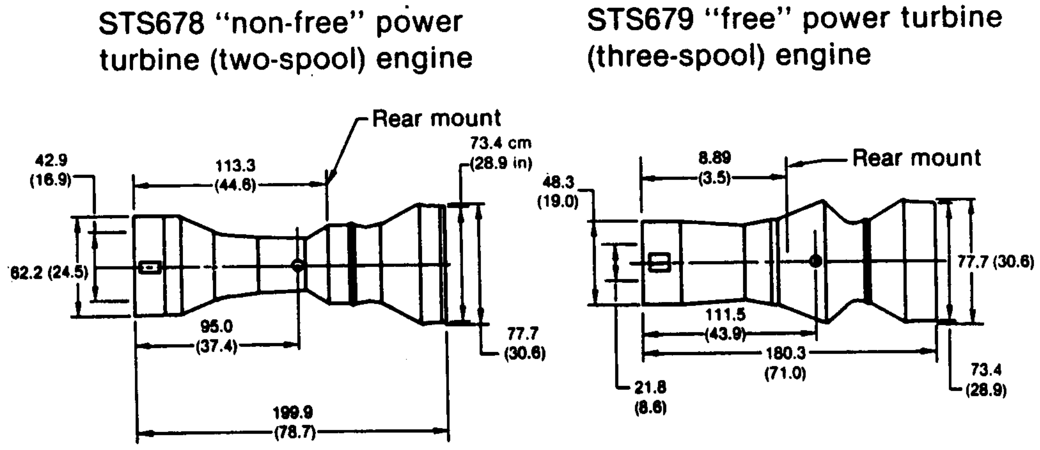


Figure 4.3-48 Base Size Turboprop Engine Dimensions - Detailed information on engine dimensions is included in the Propulsion System Integration Package. (J27638-31)

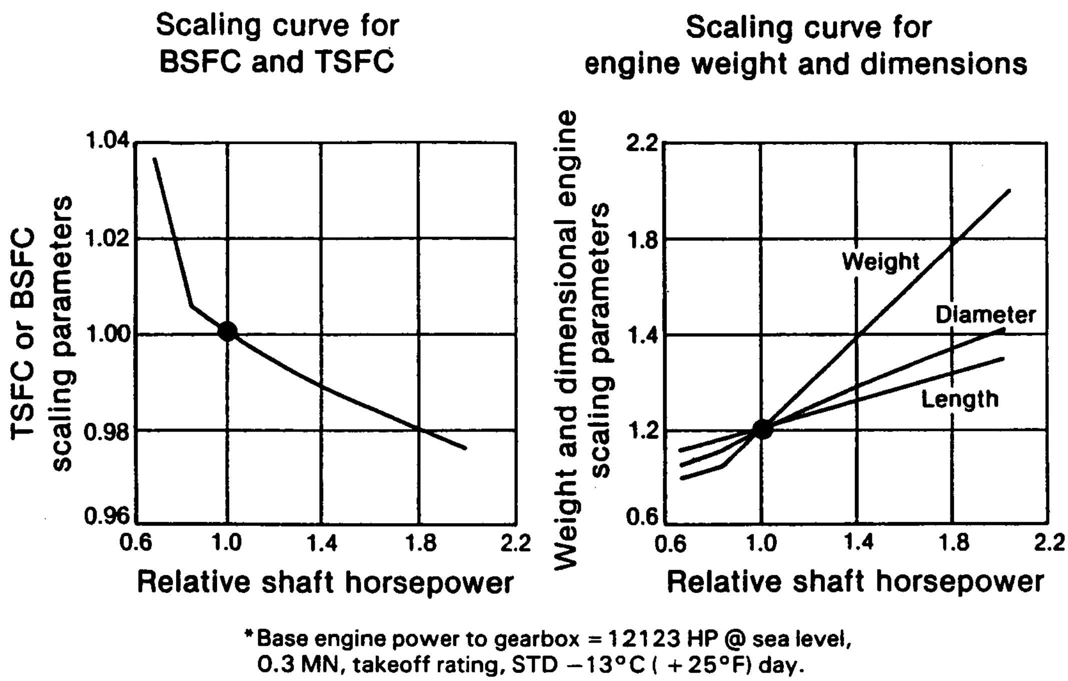


Figure 4.3-49 Turboprop Engine Scaling Information - Detailed scaling information is included in the Propulsion System Integration Package. (J27638-32)

Base size gearbox dimensions are presented in Figures 4.3-50 for the single rotation offset compound idler and in-line split path reduction gear systems. Because of the interest in counter rotation, the differential planetary counter rotation gearbox is also included in the figure. The weights of the base size engine, gearbox, and Prop-Fan are presented in Table 4.3-XXXII. Prop-Fan weight data was provided by Hamilton Standard. Weight and dimensional reduction gear scaling information is provided in Figure 4.3-51. Reduction gear performance is not expected to differ for the range of gearbox sizes considered. Scaling curves for the Prop-Fan should be obtained directly from Hamilton Standard. Base size information is provided to be consistent with the performance obtained from the computer deck included in the Propulsion System Integration Package.

4.3.4.4 Reference Turbofan Engine Propulsion System Data Package and Computer Deck (User Manual)

A comprehensive data package and computer deck were prepared for the reference turbofan engine. The capabilities and options of this package are presented in Table 4.3-XXXIII. The base size (86,072 N (19,350 lb) thrust) turbofan dimensions are shown in Figure 4.3-52. A propulsion system installation incorporating the reference turbofan is shown in Figure 4.3-53. Scaling curves for engine weight, dimensions, and performance are provided in Figure 4.3-54. This information is used to compare the capabilities of the Prop-Fan and turbofan propulsion systems.

74,662 newton-meters (55,068 ft-lb) for single rotation and
74,493 newton-meters (54,943 ft-lb) for counter rotation

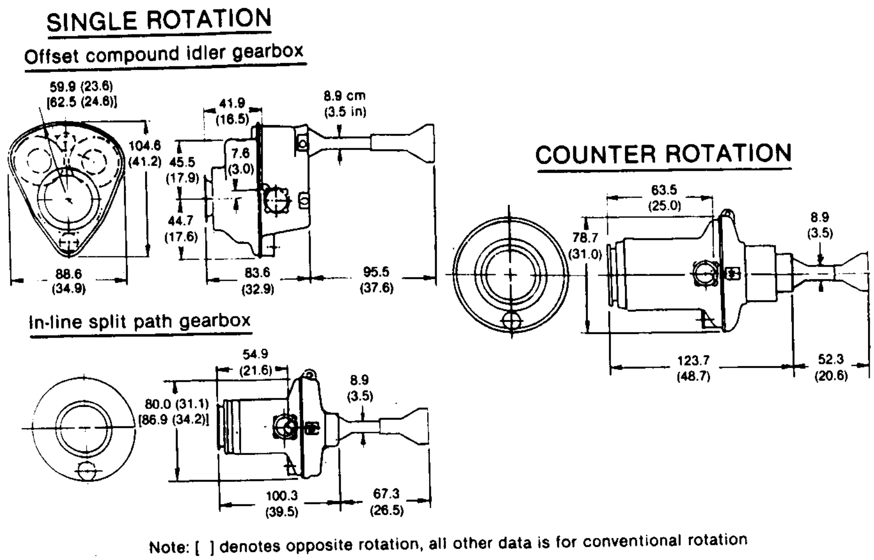


Figure 4.3-50 Base Size Gearbox Dimensions - Information is presented for both single and counter rotation gearboxes. (J27638-33)

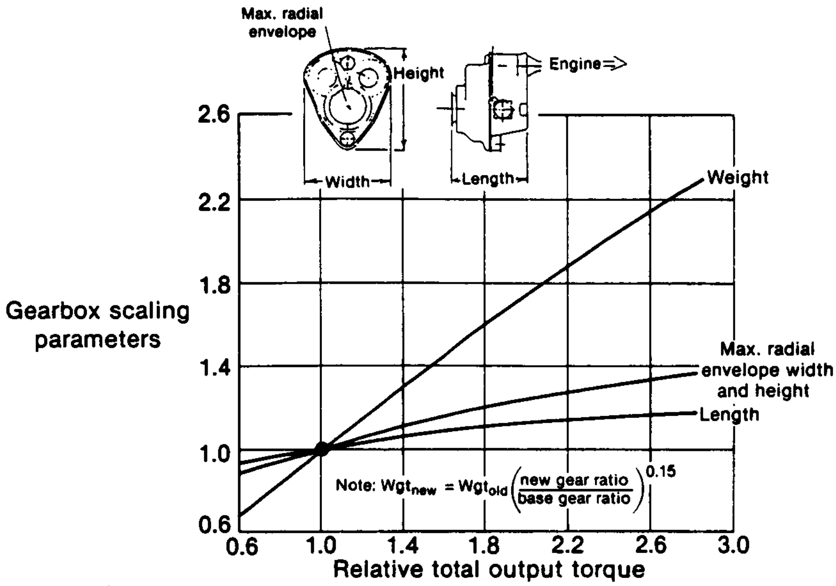
TABLE 4.3-XXXII
BASE SIZE ENGINE, GEARBOX, AND PROP-FAN WEIGHTS

	Two-Spool All Axial Non-Free Turbine STS678	Three-Spool Axial/Centrifugal Free Turbine STS679
Engine Weight, kg (lb)	839 (1850)	916 (2020)

Single and Counter Rotation Gearbox Weight, kg (lb)

	Single Rotation	
	Conventional Rotation	Opposite Rotation
o Offset Compound Idler	467 (1030)	549 (1210)
o In-Line Split Path	347 (765)	347 (765)
	Counter Rotation	
o In-Line Differential Planetary	388 (855)	

	Single Rotation (Ten Blades)	Counter Rotation (Twelve Blades)
Prop-Fan Weight, kg (lb)	671 (1480)	841 (1855)



Base size gearbox output torque @ sea level takeoff power
STD. −13°C (+25°F) day, Mach 0.3 = 74,662 newton-meters (55,068 ft-lb) for single rotation and 74,492 newton-meters (54,943 ft-lb) for counter rotation

Figure 4.3-51 Reduction Gear Scaling Data - Performance is not expected to differ significantly over the range of gearbox sizes evaluated. (J27638-34)

TABLE 4.3-XXXIII
REFERENCE TURBOFAN (STF686) COMPUTER DECK CAPABILITIES AND OPTIONS

- o Capability for user to specify inlet and/or exhaust system as desired.
- o Standard customer bleed and power extraction options.
- o Capability to schedule environmental control system bleeds with flight conditions.
- o Capability to operate as a stand-alone program or as a subroutine to the users program.
- o Capability to automatically sequence the input to allow rapid generation of performance information at a series of flight conditions.
- o User Manual provides weights, geometric dimensions, and scaling curves for the engine.

4.3.4.5 Acquisition and Maintenance Cost Information for the Turboprop and Reference Turbofan Engines

Complete acquisition and maintenance cost information, including a description of the system used to generate the cost estimates, is provided in the proprietary cost document.

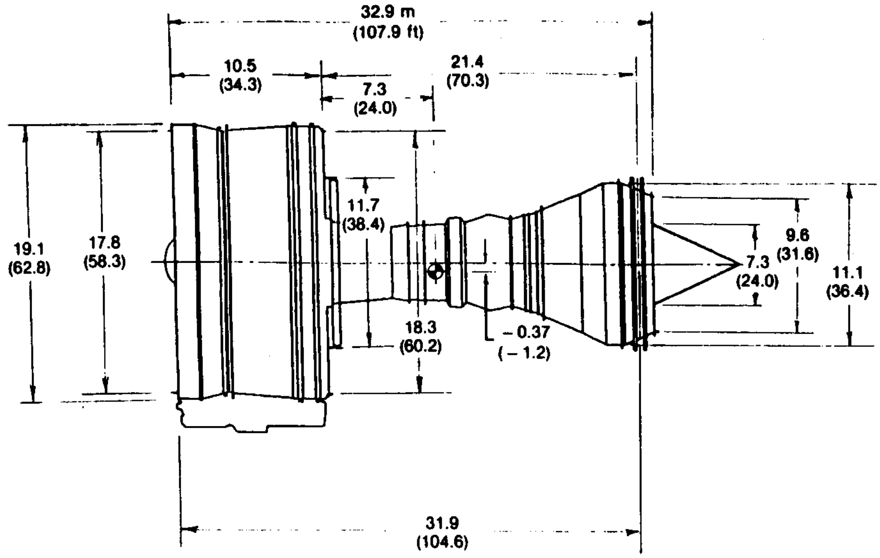


Figure 4.3-52 Reference Turbofan Engine (STF686) Dimensions - The dimensions are shown for the base size engine. (J27638-35)

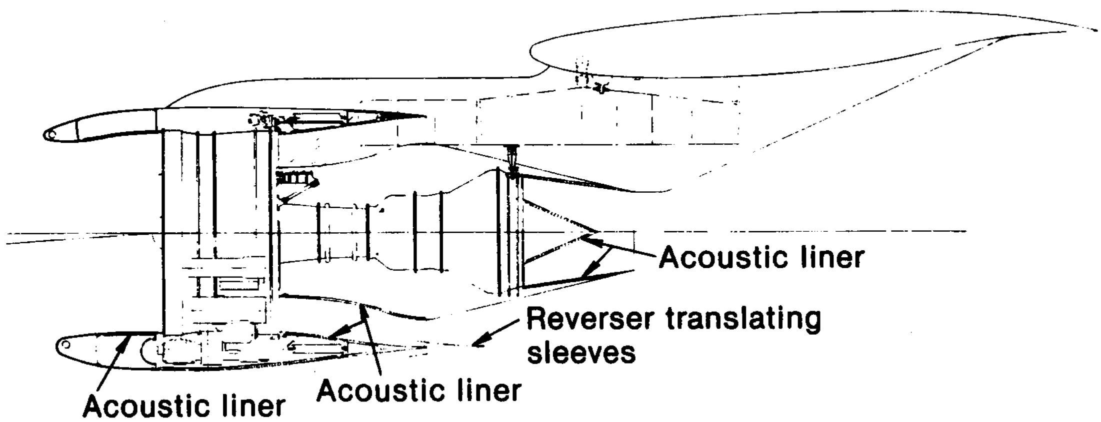


Figure 4.3-53 Reference Turbofan Engine (STF686) Installation - Acoustic liner locations and reverser concepts are highlighted. (J27638-124)

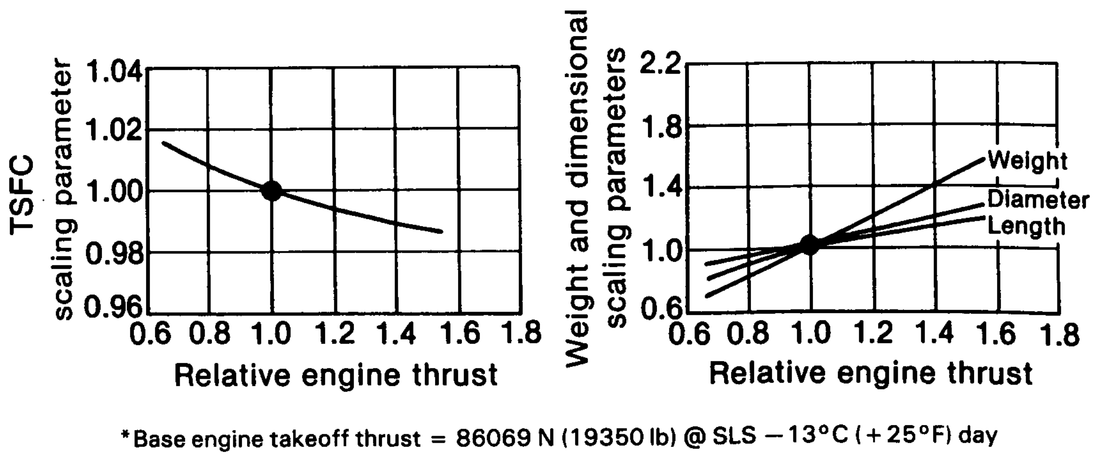


Figure 4.3-54 Reference Turbofan Engine (STF686) Scaling Information - The impact of changes in engine thrust rating on TSFC, weight, diameter, and length are shown. (J27638-37)

SECTION 4.4 -- DISCUSSION OF RESULTS
Task IV -- Engine/Aircraft Evaluation

4.4 TASK IV - ENGINE/AIRCRAFT EVALUATION

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4.4 TASK IV - ENGINE/AIRCRAFT EVALUATION

4.4.1 Introduction

The objective of Task IV was to assess the merits of the Prop-Fan propulsion system relative to a turbofan propulsion system with comparable technology. The two propulsion systems were evaluated in the reference 120-passenger Mach 0.75 cruise airplane over a simulated flight cycle covering a typical mission and the design range for the aircraft. The evaluation included airplane and engine sizing, mission performance, fuel burn, noise, and economics. The Prop-Fan powered aircraft demonstrated a 21% advantage in fuel burn and a 10% advantage in direct operating costs over the turbofan.

4.4.2 Evaluation Procedure

4.4.2.1 Evaluation Ground Rules

Ground rules for the evaluation were established in Task I; study procedures and assumptions are summarized in Section 4.1. Several of the ground rules which are especially important to the comparison of the Prop-Fan propulsion system to the reference turbofan are restated below:

- Cruise speed of Mach 0.75.
- Design range of 3333 km (1800 nm), typical mission range of 740 km (400 nm).
- Engine sizing requirements of
  - Federal Aviation Regulation (FAR) takeoff field length of 2133 m (7000 ft) at sea level, 28°C (84°F) day;
  - Initial cruise altitude capability of 9448 m (31,000 ft) or 10,668 m (35,000 ft) on design mission.
- No drag penalty for propeller slipstream swirl effect on wing.
- Fuselage acoustic treatment weight penalty added to Prop-Fan powered airplanes to reduce cabin noise to turbofan levels.

Cruise speed affects the comparison because the performance advantage of the Prop-Fan powered aircraft increases as Mach number decreases. Mach 0.75 was chosen, after consultation with the airframe manufacturers as representative for airplanes of this size (120 passengers) and range capability. The effect of range on the comparison is a bit more subtle. At short ranges - less than 926 km (500 nm) - a large portion of the mission fuel is burned at lower Mach numbers, where the Prop-Fan powered aircraft has relatively better performance. At mid-ranges (1852 km (1000 nm) to 3704 km (2000 nm)) mission fuel is dominated by cruise, so the fuel burn advantage of the Prop-Fan powered aircraft approaches its cruise TSFC advantage, which is lower than at lower Mach numbers. At long ranges - greater than 4630 km (2500 nm) - cruise fuel burn is of course the dominant factor in mission fuel, but the growth of airplane fuel fraction (fuel weight/gross weight) gives the fuel efficiency of the Prop-Fan powered aircraft increasing leverage on airplane empty weight.

Thus the fuel burn advantage of the Prop-Fan powered aircraft will tend to reach a minimum at mid ranges and increase at both shorter and longer ranges. The design and typical mission ranges chosen for evaluation are representative of current airplanes in this class.

Engine sizing requirements affect the Prop-Fan propulsion system/turbofan comparison through the differing thrust lapse rates with Mach number of the two systems. The thrust of the turboprop propulsion system lapses faster than the thrust of the turbofan with increasing Mach number. Therefore, the Prop-Fan propulsion system is relatively more sensitive to cruise sizing than the turbofan, which is why two cruise altitude requirements were chosen for evaluation.

The effect of propeller slipstream swirl on wing aerodynamics has not yet been established. Wind tunnel tests run at NASA-Ames have suggested that, with proper wing shaping, some of the slipstream swirl energy may be recoverable, which provide a potential drag benefit for the Prop-Fan propulsion system. Much more work, both experimental and analytical, is needed before any penalty (or benefit) can be definitively assessed. In the baseline comparisons of the APET study, no penalty or benefit is assumed for swirl/wing interaction. The effects of Prop-Fan drag penalty on the comparison will be assessed separately in Section 4.4.5.

The amount of cabin acoustic treatment required on a Prop-Fan powered airplane to achieve interior noise levels comparable to a turbofan powered airplane is another unknown in the comparison. Preliminary studies conducted by industry and NASA suggest that the penalty of 1.7% of maximum takeoff gross weight predicted for this study may be conservative. The effect of cabin treatment weight on the comparison of the Prop-Fan powered aircraft and the turbofan powered aircraft will be evaluated in Section 4.4.5.

4.4.2.2 Airplane Sizing - Mission Analysis Procedure

The airplane sizing/mission analysis procedure used in the APET Program is diagrammed in Figure 4.4-1. Since this is a "rubber" airplane analysis, several loops are needed to solve for the airplane size takeoff gross weight required to perform the design mission. A derivative program created by the Vehicle Analysis Modular Programming System (VAMP) was used to perform the complete process outlined in the figure. Output includes flight conditions and thrust requirements for Federal Aviation Regulation Part 36 noise calculations.

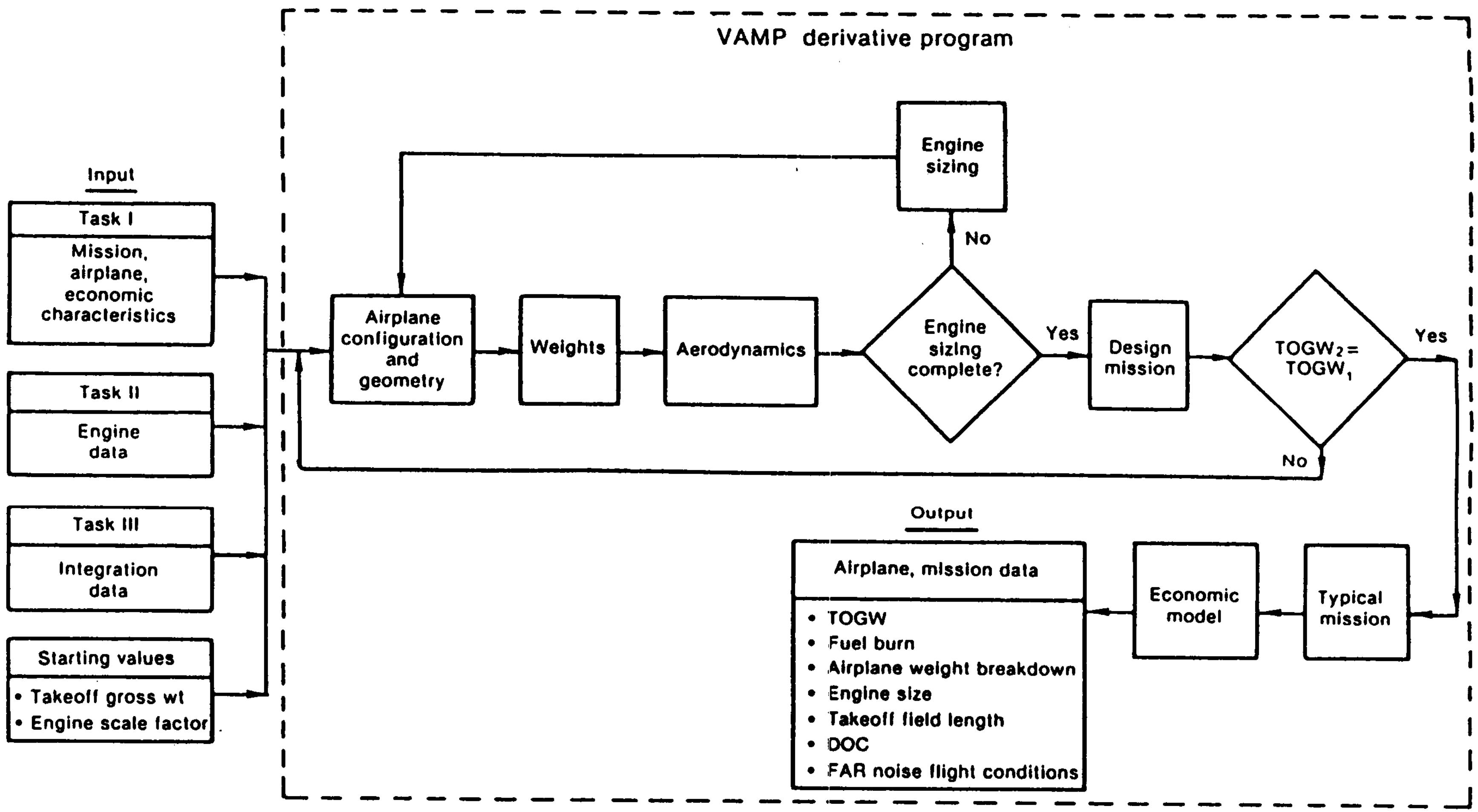

Figure 4.4.1 Mission Analysis Procedure - The Vehicle Analysis Modular Programming System was used for airplane sizing and mission analysis. (J27638-83)

4.4.3 Turbofan/Prop-Fan Propulsion System Comparison

To conduct the engine/aircraft evaluation, one of the Prop-Fan propulsion systems from Task III was selected for mission evaluation. The system consists of the two-spool axial compression engine (STS678) with an offset compound idler gearbox and single scoop inlet. The system was wing mounted, so opposite rotation was used. (Propellers on opposite sides of the airplane turn in opposite directions; therefore, both wings encounter the same slipstream characteristics.) Even though there are penalties in gearbox weight, price, maintenance and efficiency with opposite rotation, the potential advantage in wing aerodynamics makes the system desirable. Since opposite rotation caused differences in performance and other characteristics between right and left engines on the airplane, averaged values were used in the mission analysis. These averaged values are also used in the various comparisons between the turbofan and Prop-Fan propulsion systems in this part of the report.

In this section, engines will often be referred to as "base size" or "scaled to airplane size." The first term indicates that the engines are in their unscaled, field deck thrust (or shp) size (86,295 N (19,350 lb) thrust at sea level standard day, static takeoff for the turbofan; 12,000 shp at sea level standard day, Mach 0.2 takeoff for the turboprop). The second designation is applied to engines which have been scaled as required to perform the design mission, with the payload, range, and sizing requirements specified in the study procedures and assumptions (Section 4.1). The 10,668 m (35,000 ft) initial cruise altitude, 2133 m (7000 ft) takeoff field length case was used for comparison.

4.4.3.1 Performance Comparison

Performance of the STS678 Prop-Fan propulsion system at a cruise speed of Mach 0.75 and a cruise altitude of 10,668m (35,000 ft) is compared to the performance of the STF686 reference turbofan engine in Figure 4.4-2. Performance shown in the figure includes the effects of customer horsepower extraction, real inlets and nozzles and propeller pressure rise. Nacelle drag, both free stream and scrubbing, is not included. Data are in airplane size. The Prop-Fan powered aircraft has 16% better specific fuel consumption at the maximum cruise rating.

The performance advantage of the Prop-Fan powered aircraft increases dramatically as flight Mach number is reduced. Figure 4.4-3 shows that at sea level takeoff, Mach 0.2, the TSFC advantage of the Prop-Fan powered aircraft has risen to 39%. This difference in performance lapse with flight speed is caused by the specific thrust (thrust/airflow) difference between the turbofan and turboprop engines. The turbofan has a higher specific thrust than the turboprop, and, hence, the performance of the turbofan does not improve as rapidly as the performance of the turboprop as speed is reduced.

Isolated pod installation effects at cruise are compared in Table 4.4-I. Pressure rise through the propeller offsets the higher inlet loss of the Prop-Fan propulsion system. Nacelle drag favors the Prop-Fan system, due primarily to a somewhat smaller surface area and a lack of high velocity fan exhaust scrubbing drag. Also contributing are a better length to diameter ratio and elimination of mismatches associated with thrust reversers.

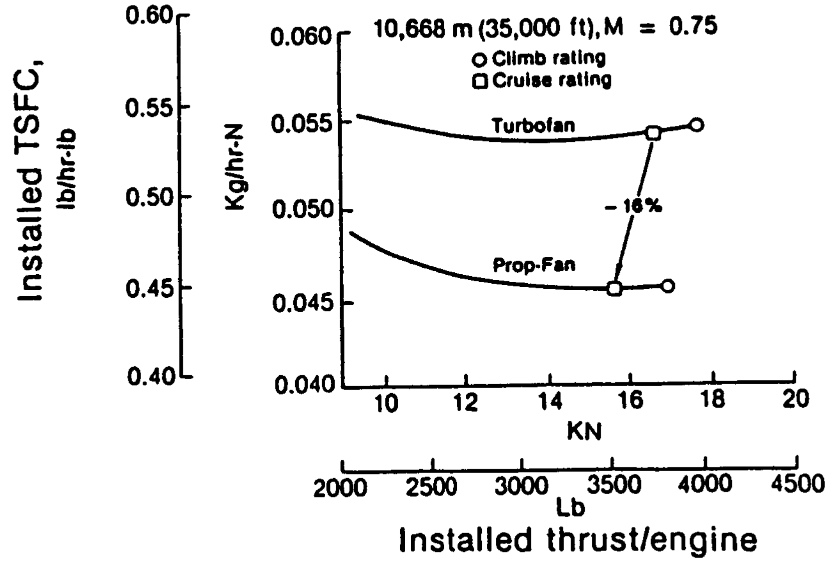

Figure 4.4-2 Cruise Performance Comparison - The turboprop engine has more than 16% better thrust specific fuel consumption at maximum cruise than the reference turbofan. Both engines are shown in airplane size,and include horsepower extraction, real inlets and nozzles. (J27638-72)

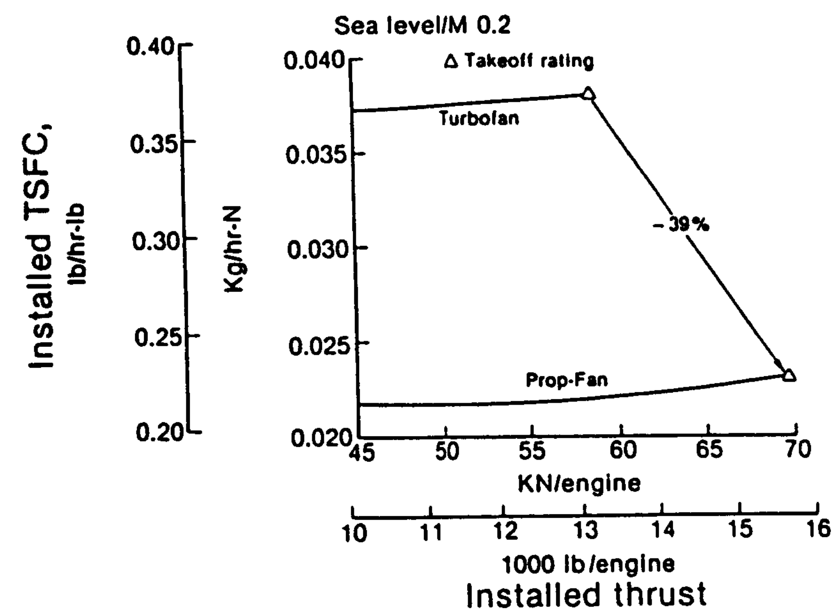

Figure 4.4-3 Takeoff Performance Comparison - The performance advantage of the turboprop engine at takeoff is much greater than the advantage seen at cruise. The thrust specific fuel consumption of the turboprop engine is 39% lower than the TSFC of the turbofan at the takeoff rating. Both engines are shown in airplane size, and include horsepower extraction, real inlets and nozzles. (J27638-73)

TABLE 4.4-I
ISOLATED POD INSTALLATION EFFECTS

Mach 0.75, 10,668 m (35,000 ft)
Airplane Size Engines

	Base Units		Percent Change in TSFC	
	Prop-Fan	Turbofan	Prop-Fan	Turbofan
Propeller Inlet Pressure Rise (% P/P)	3.0	---	-1.24	---
Inlet Pressure Loss (% P/P)	3.0	0.4	+1.24	+0.57
Fan Duct Pressure Loss (% P/P)	---	1.27	---	+1.02
Primary Exhaust Pressure Loss (% P/P)	1.0	0.79	+0.35	+0.27
Fan Nozzle C_V	---	0.995	---	+1.00
Primary Nozzle C_V	0.965	0.996	+0.74	+0.17
Customer Power Extraction, kw (shp)	125 (168)	125 (168)	+2.75	+1.65
Isolated Nacelle Drag, N (lb)	725 (163)	1067 (240)	+4.7	+6.6
Total Isolated Pod Installation Effect			+8.54%	+11.28%

4.4.3.2 Weight Comparison

The weight comparison in Table 4.4-II shows both base size and airplane size propulsion systems. Reference thrust for the turbofan is uninstalled (no bleed or horsepower extraction, but real inlets and nozzles, no scrubbing drag) takeoff thrust at sea level static, +14°C (+25°F) day, while turboprop shaft horsepower is the uninstalled takeoff rating at sea level, Mach 0.3, +14°C (+25°F) day. Engine weight reflects the relatively small size of the gas generator in the Prop-Fan propulsion system. Gearbox weight is presented for a compound idler offset gearbox, including one half the weight difference for opposite rotation, as explained above and the engine/gearbox connecting shaft.

Nacelle weights include pylon, engine build up (engine and airframe accessories commonly located in the nacelle), and, for the turbofan, thrust reverser weights. The airplane size Prop-Fan propulsion system is slightly heavier than the turbofan propulsion system, primarily due to gearbox and propeller weight.

TABLE 4.4-II
TURBOFAN/TURBOPROP COMPARISON

Installed Weight

	Base Size		Airplane Size	
	Turbofan	Prop-Fan	Turbofan	Prop-Fan (10,668 m)
Size (thrust at sea level static, shp at sea level, Mach 0.3, STD + 14°C)	86,069N (19,350 lb)	8948 kw (12,000 hp)	73,926 N (16,620 lb)	8620 kw (11,560 hp)
Engine Weight, kg (lb)	1588 (3500)	839 (1850)	1384 (3051)	811 (1787)
Propeller Weight, kg (lb)	---	671 (1480)	-	640 (1411)
Offset Gearbox Weight, kg (lb)	---	531 (1170)	-	509 (1123)
Nacelle Weight, kg (lb)	1263 (2785)	805 (1775)	1116 (2461)	782 (1723)
Total Weight, kg (lb)	2851 (6285)	2846 (6275)	2500 (5512)	2742 (6044)

4.4.3.3 Cost Comparison

The maintenance cost and acquisition price of the two systems are compared in Table 4.4-III. The lower price of the turboprop engine is due to smaller size and lack of a fan. These factors carry over into maintenance cost, where a similar effect can be seen. Addition of a gearbox and propeller (Prop-Fan) brings the total price of the Prop-Fan propulsion system to within 10% of the price of the turbofan, while addition of a reverser restores the 17% difference. The effect of the gearbox and propeller on maintenance cost is somewhat less. The relatively low maintenance cost of the gearbox is due to the compound idler design, with few gears and bearings, and to the advanced technologies which are assumed to be available. If the split path inline gearbox had been used, maintenance cost would have been almost double the cost of the compound idler offset gearbox. Propeller price and maintenance cost were obtained from the Hamilton Standard Division of United Technologies Corporation.

TABLE 4.4-III
COST COMPARISON SCALED TO AIRPLANE SIZE

	Acquisition		Maintenance	
	STF686 Turbofan	STS678 Prop-Fan (10,668m)	STF686 Turbofan	STS678 Prop-Fan (10,668m)
Engine	91.5%	64.2%	95.8%	68.8%
Gearbox (Offset)	---	8.7%	---	3.9%
Propeller	---	10.3%	---	4.6%
Reverser	8.5%	---	4.2%	---
Total	100.0%	83.2%	100.0%	77.3%

4.4.4 Mission Analysis - Airplane Sizing and Performance

The results of a "fly-off" between the Prop-Fan powered aircraft and the turbofan powered aircraft are discussed in this section in terms of airplane/engine sizing, airplane characteristics, mission performance, and fuel burn.

4.4.4.1 Airplane Sizing

In Table 4.4-IV the STF686 turbofan powered airplane is compared to two Prop-Fan powered airplanes. Recalling that the airplane sizing requirements in the ground rules called for a Federal Aviation Regulation takeoff field length (TOFL) of 2133 m (7000 ft) or less and initial cruise altitude capabilities (ICAC) of 9448 m (31,000 ft) or 10,668 m (35,000 ft), it can be seen that when the turbofan is sized to meet the takeoff requirement it has sufficient cruise thrust to exceed the highest initial cruise altitude capability. The turboprop engine, due to its different thrust lapse rate with Mach number, is sized by the cruise requirements, and, in each case, has enough takeoff thrust to better the takeoff field length requirement. [The 9448 m (31,000 ft) initial cruise altitude capability can be seen to provide a good match to the 2133 m (7000 ft) field length given the performance of the Prop-Fan propulsion system and the airplane assumptions.]

TABLE 4.4-IV
AIRPLANE CHARACTERISTICS

	Turbofan	Prop-Fan	
		9448 m	10,668 m
Design Range, km (Nm)	3333 (1800)	3333 (1800)	3333 (1800)
Payload - 120 Passengers, kg (lb)	10,886 (24,000)	10,886 (24,000)	10,886 (24,000)
Cruise Mach Number	0.75	0.75	0.75
Operating Empty Weight, kg (lb)	32,420 (71,470)	32,390 (71,410)	33,830 (74,590)
Maximum Takeoff Gross Weight, kg (lb)	52,680 (116,100)	50,970 (112,400)	52,480 (115,700)
Initial Cruise Altitude Capability, m (ft)	11,064 (36,300)	9448 (31,000)	10,668 (35,000)
Takeoff Field Length at Maximum Takeoff Gross Weight, m (ft) (S.L., Std + 14°C)	2133 (7000)	2130 (6990)	1639 (5380)
Reference Wing Area, m^2 (ft^2)	103 (1106)	91 (977)	98 (1052)
Cruise Lift-to-Drag Ratio	16.49	16.21	16.52
Engine Size Static Fn, Mach 0.3 SHP, at S.L. Std + 14°C	73,926 N (16,620 lb)	7405 kw (9930 hp)	8620 kw (11,560 hp)
Propeller Diameter, m (ft)	--	3.70 (12.14)	3.99 (13.10)

The primary cause of the differences between the two Prop-Fan powered aircraft is the reduction in engine size which results from decreasing initial cruise altitude capability from 10,668 m (35,000 ft) to 9448 m (31,000 ft). This 14% reduction in engine size leads to a 4% decrease in operating empty weight and a 3% reduction in takeoff gross weight. Since the takeoff field length for the Prop-Fan powered aircraft sized for 9448 m (31,000 ft) is very close to 2133 m (7000 ft), no further reduction in engine size would be permitted under the study ground rules, unless the Prop-Fan engine rating schedule was changed. Wing loading (gross weight/wing area) was chosen to minimize fuel burn on a 740 km (400 nm) typical mission, subject to an upper limit of 115 lbs/ft^2 at design maximum takeoff gross weight.

The enroute engine-out cruise altitude capabilities of the turbofan powered airplane and the two Prop-Fan powered airplanes are considerably different. Although this factor was not considered a sizing condition due to the relatively small size and short range of the aircraft, it does provide an interesting comparison of the performance of the three airplanes. Engine out capabilities were determined using maximum continuous thrust ratings for the "begin cruise" weight of a 1852 km (1000 nm) mission with 100% and 60% load factors. With a 100% load factor, the turbofan powered airplane could sustain an altitude of 5580 m (18,300 ft) with an engine out, while the Prop-Fan powered aircraft sized for 10,668 m (35,000 ft) could sustain an altitude of 4600 m (15,100 ft) and the Prop-Fan powered aircraft sized for 9448 m (31,000 ft) demonstrated a 3380 m (11,100 ft) capability. With a 60% load factor, the enroute engine out capability of the turbofan powered airplane was 6280 m (20,600 ft), while the capability of the Prop-Fan powered aircraft sized for 10,668 m (35,000 ft) was 5460 m (17,900 ft) and the capability of the Prop-Fan powered aircraft sized for 9448 m (31,000 ft) was 4540 m (14,900 ft). However, there is some latitude available in the maximum continuous ratings for the Prop-Fan powered aircraft. Thus, if enroute engine out capability was considered a critical sizing condition, the rating could be increased to help satisfy the requirement.

Comparing the turbofan powered aircraft with the Prop-Fan powered aircraft sized for 10,668 m (35,000 ft), most of the difference in operating empty weight is due to differences in the weight of the propulsion system and cabin acoustic treatment. The maximum gross weight of the turbofan is higher because it requires a larger fuel load to accomplish the design mission.

The comparison is continued in Table 4.4-V, where a detailed weight breakdown is presented for all three aircraft. Most of the differences in operating empty weight are in the areas of engine installation and cabin noise treatment. Baseline cabin noise treatment for the turbofan powered airplane is included in the weight of the fuselage and furnishings. Fuel weights are total fuel on board for the design mission, including Air Transport Association domestic reserves.

Airplane configurations for the turbofan powered aircraft and the Prop-Fan powered aircraft sized for 10,668 m (35,000 ft) are shown in Figures 4.4-4 and 4.4-5. These figures are strictly conceptual in nature and are included only to illustrate engine locations and assumed aircraft geometry. Some reshaping of the Prop-Fan powered airplane wing and tailoring of the nacelles would be required to achieve low interference drag. However, refinements of this sort are beyond the scope of the APET study.

TABLE 4.4-V
AIRPLANE WEIGHT BREAKDOWN (English Units)

		Turbofan Powered Aircraft	Prop-Fan Powered Aircraft (31,000 ft)	Prop-Fan Powered Aircraft (35,000 ft)
Maximum Takeoff Gross Weight (lb)	=	116,147	112,357	115,703
Maximum Ramp Weight (lb)	=	116,272	112,430	115,787
Operating Empty Weight (lb)	=	71,470	71,405	74,587
Payload Weight (lb)	=	24,000	24,000	24,000
Fuel On Board (lb)	=	20,802	17,025	17,200

	Turbofan Powered Aircraft		Prop-Fan Powered Aircraft (31,000 ft)		Prop-Fan Powered Aircraft (35,000 ft)	
Airframe Structure:	33,996		33,002		34,213	
Wing		12,291		11,461		12,109
Fuselage		13,789		14,019		14,038
Tail		3,117		2,786		2,976
Landing Gear		4,800		4,736		5,091
Aircraft Systems:	7,687		7,313		7,482	
Fuel Systems		605		519		521
Surface Controls		2,016		1,816		1,932
Hydraulic, Electric, Pneumatic Equipment		2,794		2,726		2,765
Air Conditioning		1,238		1,238		1,238
Anti-Icing		285		266		277
Auxiliary Power Unit		749		749		749
Engine Installation:	11,025		10,390		12,089	
Engine Weight		3,051		3,670		4,321
Wing Installation		2,461		1,525		1,723
Electronic Systems:	2,132		2,122		2,122	
Electronics		1,521		1,521		1,521
Instruments		610		600		600
Furnishings and Equipment:	11,037		11,053		11,044	
Passenger Seats		3,780		3,780		3,780
Galley Structure		927		927		927
Other Furnishings		4,234		4,234		4,234
Emergency Equipment		505		505		505
Flight Provisions		620		620		620
Cargo Handling Equipment		742		762		750
Exterior Paint		230		226		228
Extra Cabin Noise Treatment:	0		1,929		2,062	
Manufacturers Empty Weight:	65,877		65,810		69,012	
Operator Items:	5,593		5,596		5,575	
Flight Crew and Baggage		430		430		430
Cabin Crew and Baggage		620		620		620
Unusable Fuel and Oil		152		118		119
Washing and Drinking Water		370		370		370
Toilet Water and Chemicals		86		86		86
Food and Beverage		1,106		1,106		1,106
Galley Service Equipment		560		560		560
Cabin Service Equipment		330		330		330
Emergency Equipment		560		560		560
Cargo Containers		1,377		1,414		1,393
Operator Allowances		0		0		0
Operators Empty Weight:	71,470		71,405		74,587	

TABLE 4.4-V
AIRPLANE WEIGHT BREAKDOWN (Standard International Units)

	Turbofan Powered Aircraft		Prop-Fan Powered Aircraft (9449 m)		Prop-Fan Powered Aircraft (10,668 m)	
Maximum Takeoff Gross Weight (kg) =	52,684		50,965		52,483	
Maximum Ramp Weight (kg) =	52,741		50,998		52,521	
Operating Empty Weight (kg) =	32,419		32,389		33,832	
Payload Weight (kg) =	10,886		10,886		10,886	
Fuel On Board (kg) =	9,436		7,722		7,802	
Airframe Structure:	15,420		14,970		15,519	
Wing		5575		5199		5493
Fuselage		6255		6359		6368
Tail		1414		1264		1350
Landing Gear		2177		2148		2309
Aircraft Systems:	3,487		3,317		3,394	
Fuel Systems		274		235		236
Surface Controls		914		824		875
Hydraulic, Electric, Pneumatic Equipment		1267		1237		1254
Air Conditioning		562		562		562
Anti-Icing		129		121		126
Auxiliary Power Unit		340		340		340
Engine Installation:	5,001		4,713		5,484	
Engine Weight		1384		1665		1960
Wing Installation		1116		692		782
Electronic Systems:	967		963		963	
Electronics		690		690		690
Instruments		277		272		272
Furnishings and Equipment:	5,037		5,014		5,010	
Passenger Seats		1715		1715		1715
Galley Structure		420		420		420
Other Furnishings		1921		1921		1921
Emergency Equipment		229		229		229
Flight Provisions		281		281		281
Cargo Handling Equipment		337		345		340
Exterior Paint		104		103		103
Extra Cabin Noise Treatment:	0		875		935	
Manufacturers Empty Weight:	29,882		29,851		31,304	
Operator Items:	2,537		2,538		2,529	
Flight Crew and Baggage		195		195		195
Cabin Crew and Baggage		281		281		281
Unusable Fuel and Oil		69		54		54
Washing and Drinking Water		168		168		168
Toilet Water and Chemicals		39		39		39
Food and Beverage		502		502		502
Galley Service Equipment		254		254		254
Cabin Service Equipment		150		150		150
Emergency Equipment		254		254		254
Cargo Containers		624		641		632
Operator Allowances		0		0		0
Operators Empty Weight:	32,419		32,839		33,832	

Aspect ratio	10
Sweep (¼ chord)	22 degrees
Wing area	103 m2 (1106 ft2)
Engine size	16,620 lbs fn

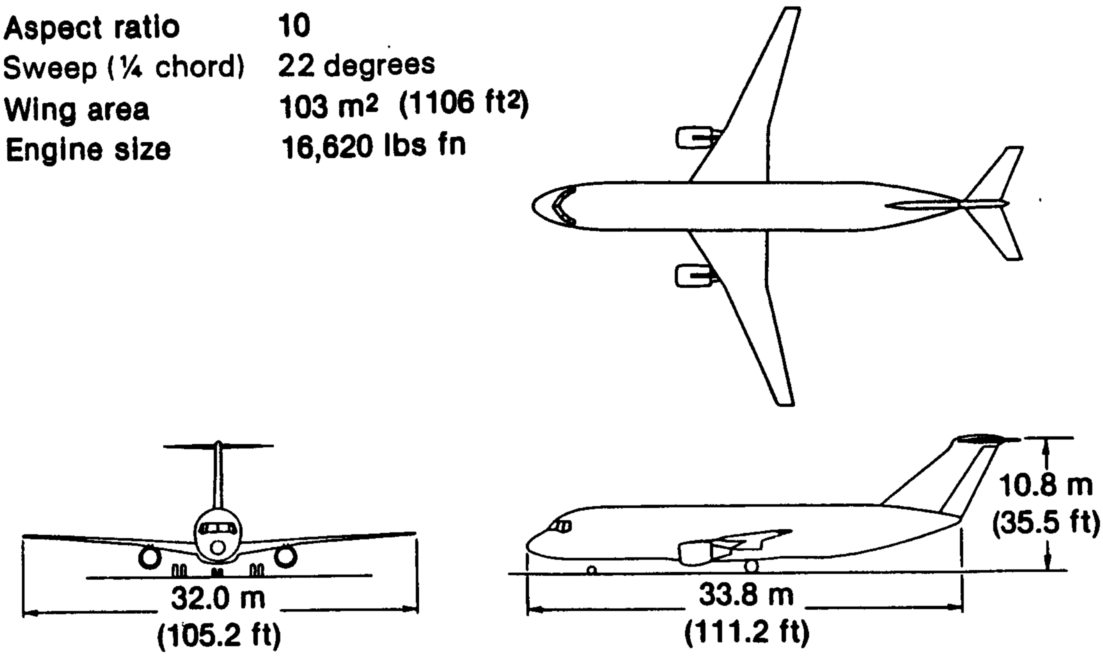

Figure 4.4-4 Turbofan Powered Airplane Configuration - This conceptual drawing shows the location of the turbofan engines and the basic airplane geometry. (J27638-214)

Aspect ratio	10
Sweep (¼ chord)	22 degrees
Wing area	98 m2 (1052 ft2)
Engine size	11,560 shp

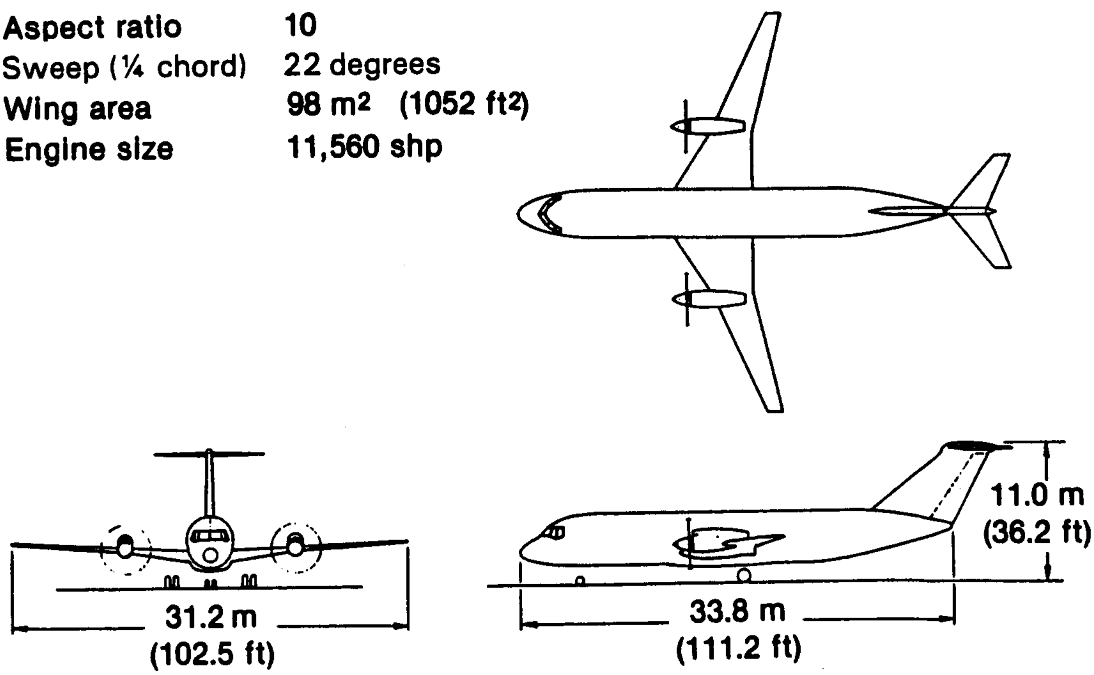

Figure 4.4-5 Prop-Fan Powered Airplane Configuration - This conceptual drawing shows the location of the Prop-Fan engines and the basic airplane geometry. (J27638-215)

4.4.4.2 Mission Performance Comparison

Fuel burn results for design and typical missions for the turbofan and Prop-Fan powered airplanes are compared in Figure 4.4-6. The fuel burn advantage for the Prop-Fan powered aircraft is 17 to 18% on the design mission and 21 to 24% on a typical mission. The advantage tends to increase as range decreases due to the relatively better low speed performance of the Prop-Fan powered aircraft. Reducing the initial cruise altitude capability requirement also increases fuel burn advantage, due primarily to the smaller engine size required at the lower altitude, as shown in Table 4.4-IV. A somewhat greater benefit is obtained from lowering the initial cruise altitude capability on a typical mission than on the design mission. This stems mostly from the small cruise lift/drag penalty assessed against the Prop-Fan powered aircraft sized for 9448 m (31,000 ft) (see Table 4.4-IV), which has more effect on the design mission which has a relatively long cruise segment.

In all cases the airplanes were allowed to cruise at the altitude at which the best fuel mileage was obtained, subject to minimum initial cruise altitude capability sizing requirements and 1220 m (4000 ft) steps (altitudes used were 9448 m, 10,668 m and 11,887 m (31,000 ft, 35,000 ft and 39,000 ft)).

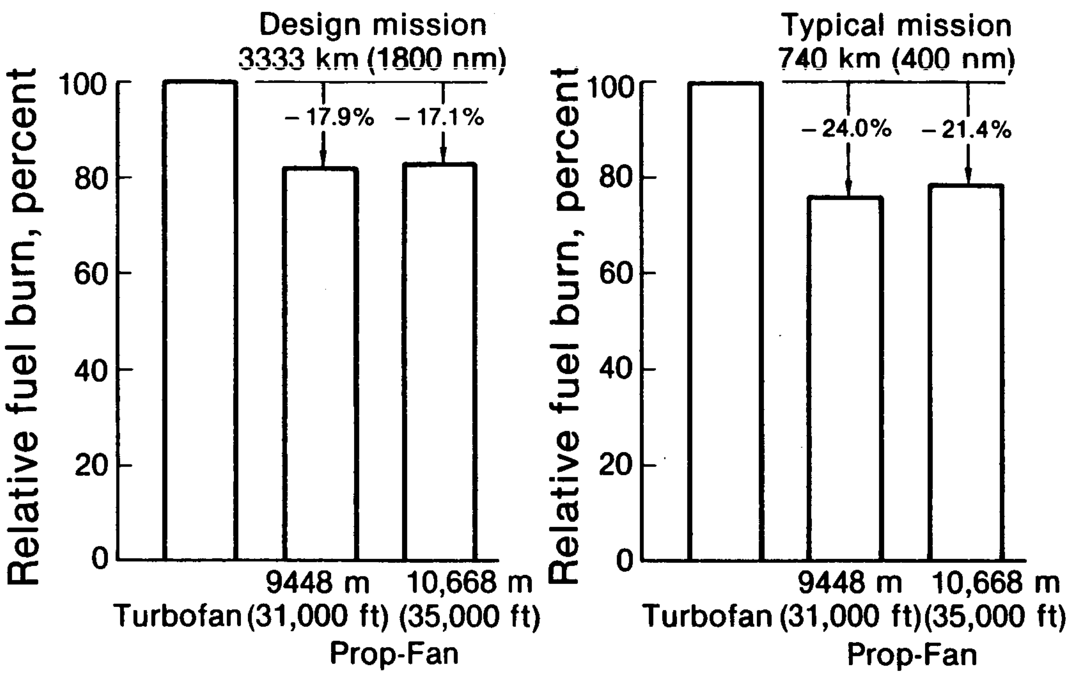

Figure 4.4-6 Fuel Burn Comparison - The Prop-Fan powered aircraft has a 17% to 18% advantage in fuel burn at the design range (3333 km) and a 21% to 24% advantage during a typical mission (740 km). (J27638-7)

Table 4.4-VI shows the increased performance advantage of the Prop-Fan powered aircraft at lower speeds. The Prop-Fan powered aircraft sized for 10,668 m (35,000 ft) demonstrates a 38% advantage in takeoff fuel use (which includes takeoff and climb to 457 m (1500 ft)) and a 14% advantage in fuel burn per mile at cruise. Cruise altitudes differ among the three aircraft due to differences in optimum conditions and thrust capabilities; the turbofan powered airplane cruises at 11,887 m (39,000 ft), for best fuel mileage while both Prop-Fan powered aircraft cruise at 10,668 m (35,000 ft) during a typical mission (see Figure 4.4-7). This makes climb performance rather difficult to compare, since the turbofan is spending a large portion of its climb at high altitudes where the airplane is more efficient. Taxi fuel, which is calculated using ground idle thrust, is another area of significant advantage for the Prop-Fan. This stems primarily from the smaller engine core size of the Prop-Fan propulsion system, which allows lower idle fuel flows. Descent performance of the Prop-Fan powered aircraft benefits from the same effect. Fuel burn at approach for the Prop-Fan powered aircraft shows a significant advantage (36%) of the same magnitude as takeoff performance.

TABLE 4.4-VI
TYPICAL MISSION FUEL BURN BREAKDOWN (English Units)

	Turbofan			Prop-Fan Sized for 31000 ft			Prop-Fan Sized for 35000 ft		
	Time min	Dist nm	Fuel lb	Time min	Dist nm	Fuel lb	Time min	Dist nm	Fuel lb
Taxi Out	9	-	125	9	-	73	9	-	84
Take Off	2	3	261	2	4	165	1	3	161
Climb	23	149	2121	24	157	1631	18	112	1393
Cruise	17	124	854	20	142	819	26	188	1117
Descent	21	124	296	17	97	127	17	97	148
Approach	4	-	154	4	-	93	4	-	100
Taxi In	5	-	69	5	-	40	5	-	47
Mission	81	400	3880	81	400	2948	80	400	3050
(Reserves)			(4910)			(3925)			(4044)

TABLE 4.4-VI
TYPICAL MISSION FUEL BURN BREAKDOWN (Standard International Units)

	Turbofan			Prop-Fan Sized for 9448 m			Prop-Fan Sized for 10,668 m		
	Time min	Dist km	Fuel kg	Time min	Dist km	Fuel kg	Time min	Dist km	Fuel kg
Taxi Out	9	-	57	9	-	33	9	-	38
Take Off	2	6	118	2	7	75	1	6	73
Climb	23	276	962	24	291	740	18	206	732
Cruise	17	230	387	20	263	372	26	348	507
Descent	21	229	134	17	180	58	17	180	67
Approach	4	-	70	4	-	42	4	-	45
Taxi In	5	-	32	5	-	18	5	-	21
Mission	81	400	1760	81	741	1337	80	741	1383
(Reserves)			(2227)			(1780)			(1834)

Further comparisons of the mission performance of the Prop-Fan powered aircraft and the turbofan powered aircraft are presented in the mission profiles of Figure 4.4-7. This figure illustrates the differences in climb performance and cruise altitude of the two Prop-Fan powered aircraft and the turbofan powered aircraft. As would be expected, the Prop-Fan powered aircraft sized for 9448 m (31,000 ft) takes the longest time to get to cruise altitude, while the Prop-Fan powered aircraft sized for 10,668 m (35,000 ft) is quickest to 10,668 m (35,000 ft), due primarily to better low speed performance. The climb speed schedule used in all cases was: 250 knots equivalent air speed (KEAS) to 3048 m (10,000 ft), 280 KEAS to Mach 0.72, Mach 0.72 to cruise altitude, accelerate at cruise altitude to Mach 0.75. The descent schedule was the reverse of climb. The 740 km (400 nm) mission includes 14 minutes of taxi time; nine minutes out and five minutes in.

Thrust and specific fuel consumption profiles for the 740 km (400 nm) typical mission are shown in Figures 4.4-8 and 4.4-9. The thrust lapse difference between the turbofan and Prop-Fan powered aircraft is illustrated graphically; the turbofan powered aircraft begins takeoff with slightly less thrust than the Prop-Fan powered aircraft sized for 9448 m (31,000 ft), but by 6096 m (20,000 ft) it has about 15% more climb thrust than the Prop-Fan powered aircraft. Cruise thrust for both Prop-Fan powered aircraft is essentially the same, since both cruise at 10,668 m (35,000 ft) on this mission.

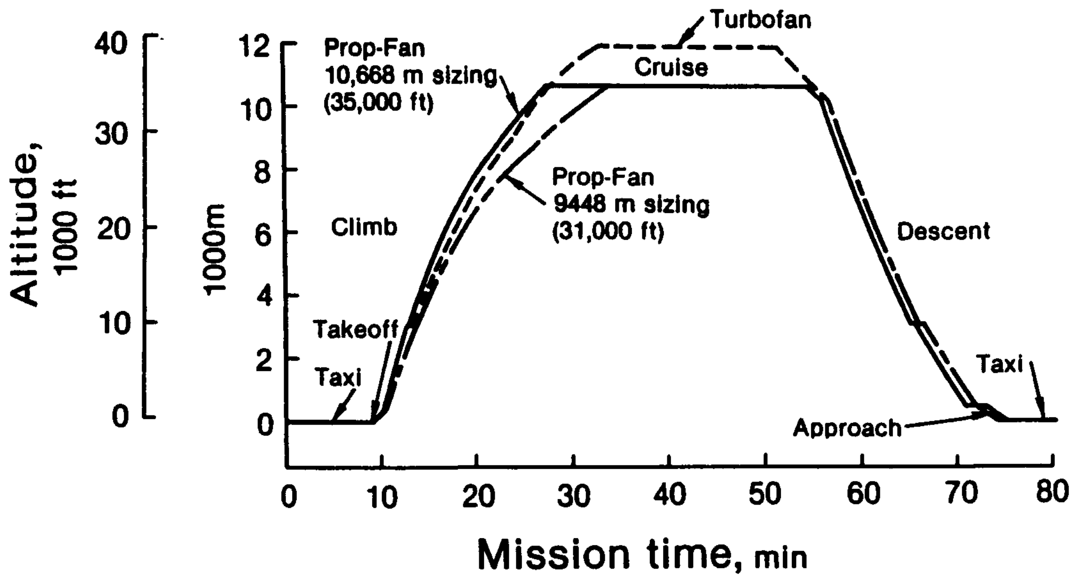

Figure 4.4-7 Typical Mission Flight Profiles - The turbofan powered aircraft cruises at a higher altitude than the Prop-Fan powered aircraft. The Prop-Fan powered aircraft sized for 9448 m has a significantly longer climbing time than the Prop-Fan powered aircraft sized for 10,668 m or the reference turbofan. (J27638-75)

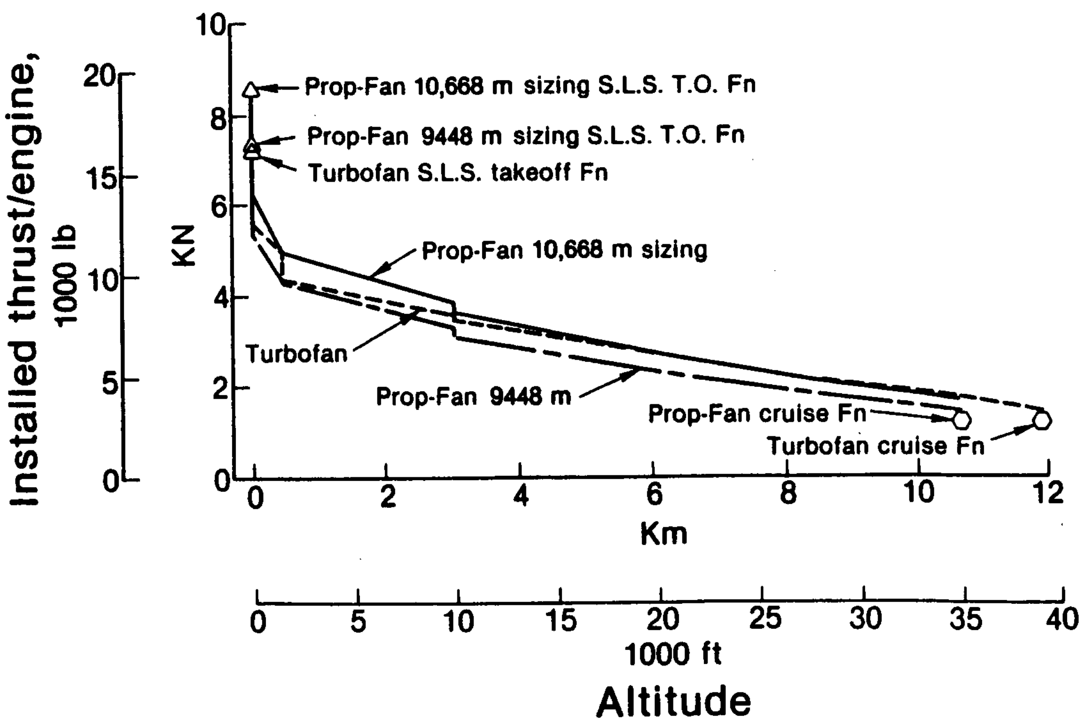

Figure 4.4-8 Typical Mission Thrust Profiles - The turbofan powered aircraft begins takeoff with slightly less thrust than the Prop-Fan powered aircraft but by 6096 m has about 15% more climb thrust than the smaller Prop-Fan. (J27638-76)

Thrust specific fuel consumption profiles in Figure 4.4-9 show that the advantage of the Prop-Fan narrows as altitude increases (actually as speed increases, but Mach number is increasing during climb up to 8686 m (28,500 ft), after which it is constant until cruise altitude is reached, so altitude and speed are roughly synonymous). Except for a minor adjustment in TSFC due to scaling effects, both Prop-Fan powered aircraft have the same thrust specific fuel consumption. The jog at 3048 m (10,000 ft) is caused by the climb speed schedule, in which there is an acceleration from 250 to 280 KEAS at that altitude. There is an additional jog in the turbofan line at 457 m (1500 ft) where the active clearance control switches on.

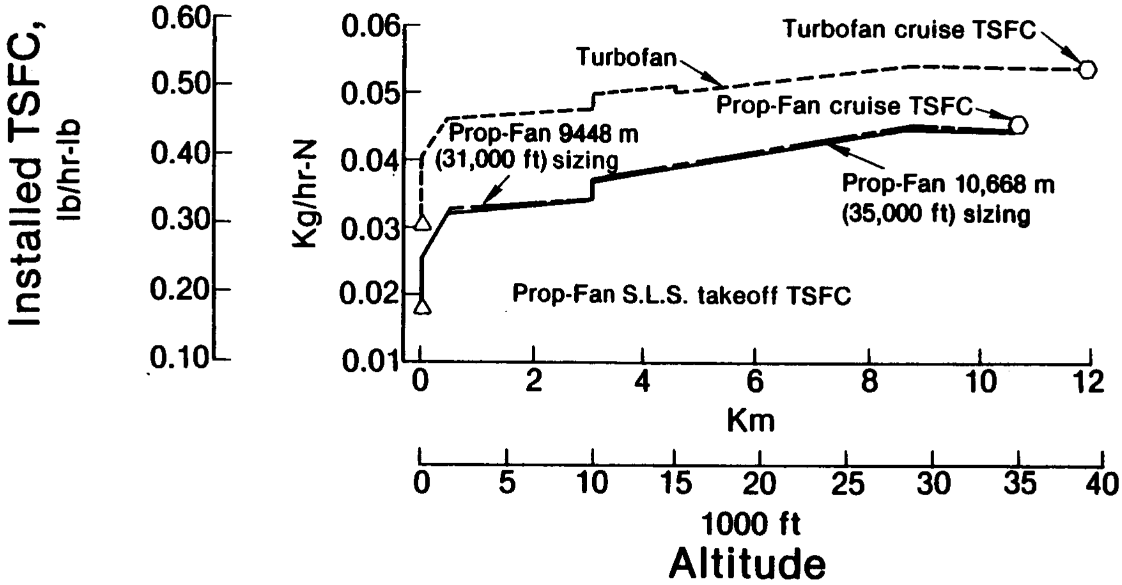

Figure 4.4-9 Typical Mission Specific Fuel Consumption Profile - The advantage in thrust specific fuel consumption demonstrated by the Prop-Fan powered aircraft increases as altitude and speed decrease. (J27638-77)

4.4.4.3 Fuel Burn Influence Coefficients

The effect of variations in engine and airplane performance parameters on fuel burn for a 740 km (400 nm) typical mission is shown in Figure 4.4-10. The Prop-Fan powered aircraft sized for 10,668 m (35,000 ft) was used as the base engine in determining these variations. Influences were calculated using the complete airplane sizing/mission analysis cycle. For example, the thrust specific fuel consumption (TSFC) influence coefficient was determined by first factoring all fuel flows in the basic, unscaled engine upward by five percent. Next, the airplane and engine were sized to perform the design mission, using the factored engine data. After all the requirements for the design mission were satisfied (cruise altitude, takeoff field length, payload, range, etc.), the resulting airplane and engine were flown through the 740 km (400 nm) typical mission to determine fuel burn and economics for that mission.

No significance should be attached to the size of the variation selected for each parameter. The magnitude of the variation was large enough to measure accurately but small enough to remain in the linear range of effect.

These influence coefficients are very similar to the coefficients developed during Task I (Study Procedures and Assumptions) for use in the Task II (Cycle and Configuration Study) and Task III (Propulsion System Integration) evaluations. Any differences arise from the fact that the coefficients developed in Task I were based on a less advanced engine.

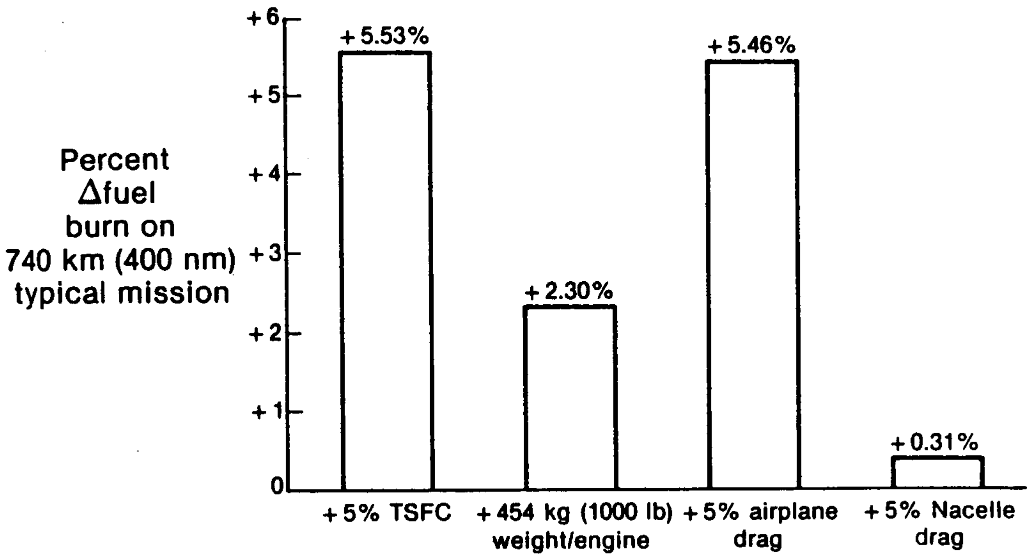

Figure 4.4-10 Prop-Fan Fuel Burn Influence Coefficients - The impact of each variation was determined by a complete airplane resizing and mission analysis. (J27638-78)

4.4.5 Mission Analysis - Economics

Direct operating cost (DOC) was selected as the figure of merit for the economic evaluation. The 1981 Boeing DOC method was furnished by The Boeing Company for use in the APET Program. Key economic ground rules included: $0.396 per liter ($1.50 per gallon) fuel price; 1981 dollars; two man crew. The ground rules for the economic evaluation are summarized in Section 4.1.6.

4.4.5.1 Direct Operating Cost Comparison

Figure 4.4-11 shows that the Prop-Fan powered aircraft offers significant reductions in direct operating cost relative to the reference turbofan: the Prop-Fan powered aircraft sized for 10,668 m (35,000 ft) is 10% less expensive to operate, while the Prop-Fan powered aircraft sized for 9448 m (31,000 ft) provides an 11.8% reduction in direct operating cost. Most of this reduction in operating cost stems from the fuel burn advantage of the Prop-Fan propulsion system, as shown in Table 4.4-VII. Reduced engine maintenance cost is also a contributing factor. Relative costs are based on cost per seat - kilometer (statute mile). Utilization is 2537 trips per year for all three aircraft.

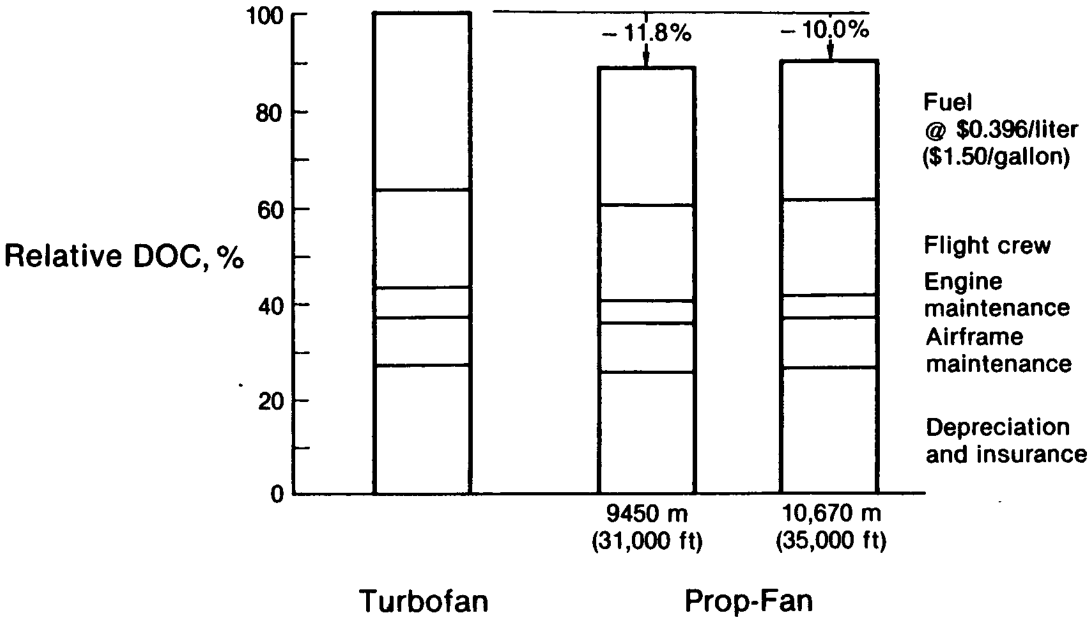

Figure 4.4-11 Direct Operating Cost Comparison - The direct operating cost of a Prop-Fan powered aircraft is 10% to 12% lower than the operating cost of a comparable turbofan powered aircraft. (J27638-79)

TABLE 4.4-VII
DIRECT OPERATING COST BREAKDOWN FOR A TYPICAL MISSION (740 km)

	STF686 Turbofan	STS678 Prop-Fan 9448 m	STS678 Prop-Fan 10,668 m
Fuel	37.0%	28.1%	29.1%
Flight Crew	20.3	20.1	20.0
Airframe Maintenance	10.3	10.1	10.3
Engine Maintenance	6.1	4.5	4.7
Insurance	1.8	1.8	1.8
Depreciation			
Engine	5.3	4.2	4.4
Airframe	19.2	19.4	19.7
Total	100.0%	88.2%	90.0%

4.4.5.2 Direct Operating Cost Influence Coefficients

The effect of variations in performance and economic factors on the direct operating cost of the Prop-Fan powered aircraft is shown in Figure 4.4-12. These coefficients were calculated in the same manner as the fuel burn coefficients (by completing the airplane sizing/mission analysis cycle), with the addition of maintenance cost and price. Unlike the fuel burn coefficients, however, direct operating cost is more sensitive to airplane drag than to thrust specific fuel consumption. The reason for this difference lies in the fact that the Prop-Fan powered aircraft is sized for cruise; thus, drag directly affects engine size, and hence, engine price and maintenance cost. Thus, while drag and specific fuel consumption have an essentially equal impact on fuel burn, the added influence of engine cost parameter scaling causes drag to have more effect on direct operating cost. All coefficients were calculated using a fuel price of $0.396/liter ($1.50/gallon).

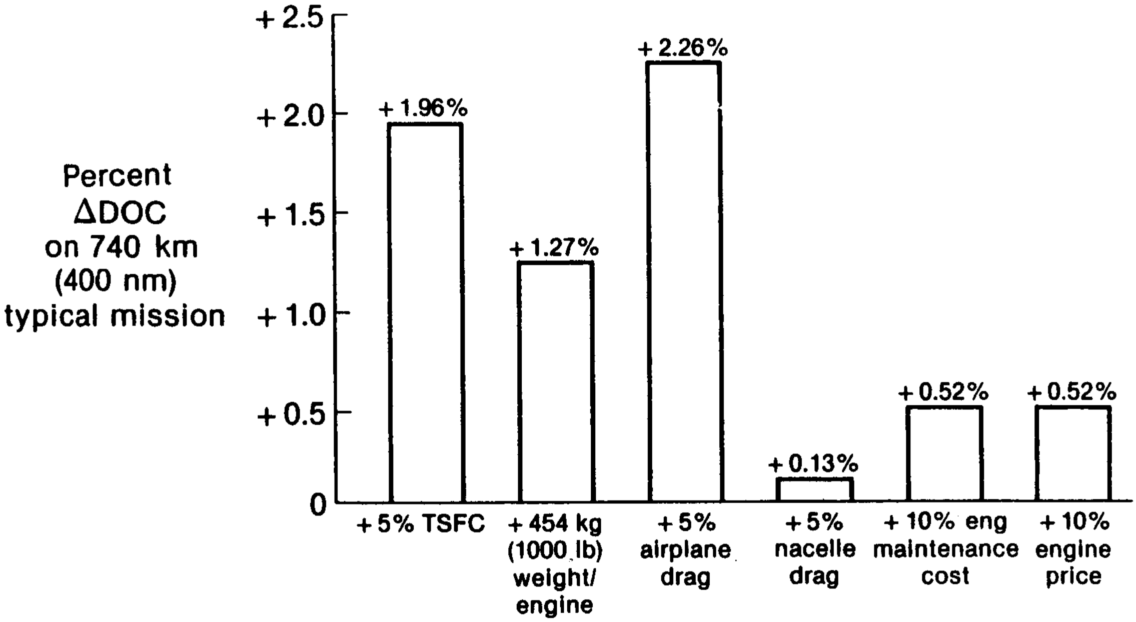

Figure 4.4-12 Direct Operating Cost Influence Coefficients - Variations in airplane drag, thrust specific fuel consumption, and engine weight have the greatest influence on direct operating cost. (J27638-80)

4.4.6 Influence of Undefined Factors on the Evaluation

As discussed earlier, there are several key parameters which may affect the performance of a Prop-Fan powered aircraft which have not yet been adequately defined. Two key factors, airplane drag and fuselage acoustic treatment weight, are addressed in Figures 4.4-13 and 4.4-14. The impact of changing fuel prices is also assessed.

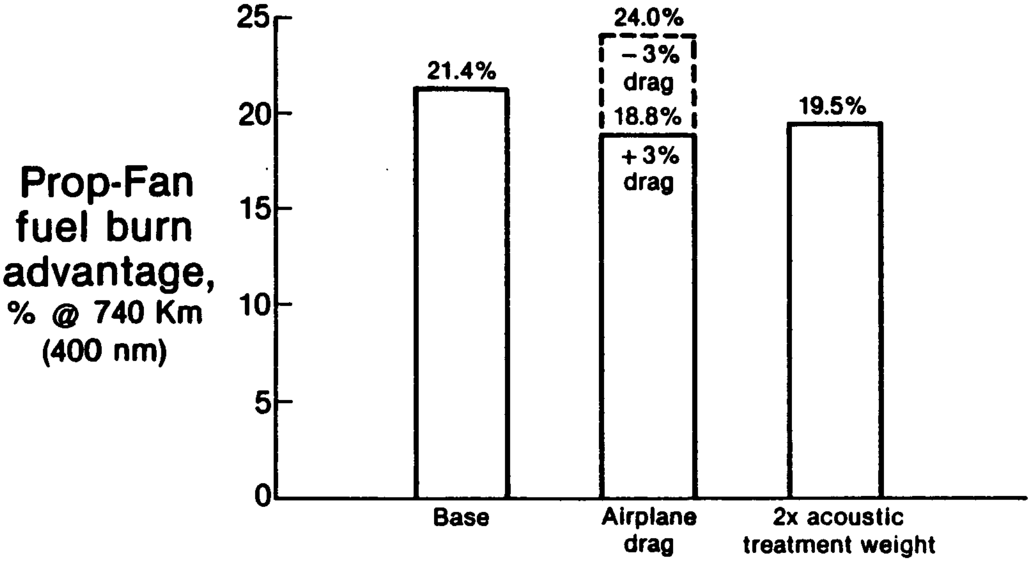

Figure 4.4-13 Effect of Key Parameters on the Fuel Burn Advantage of the Prop-Fan Powered Aircraft - Even with increased airplane drag and acoustic treatment weight, the fuel burn advantage of the Prop-Fan powered aircraft remains large. (J27638-81)

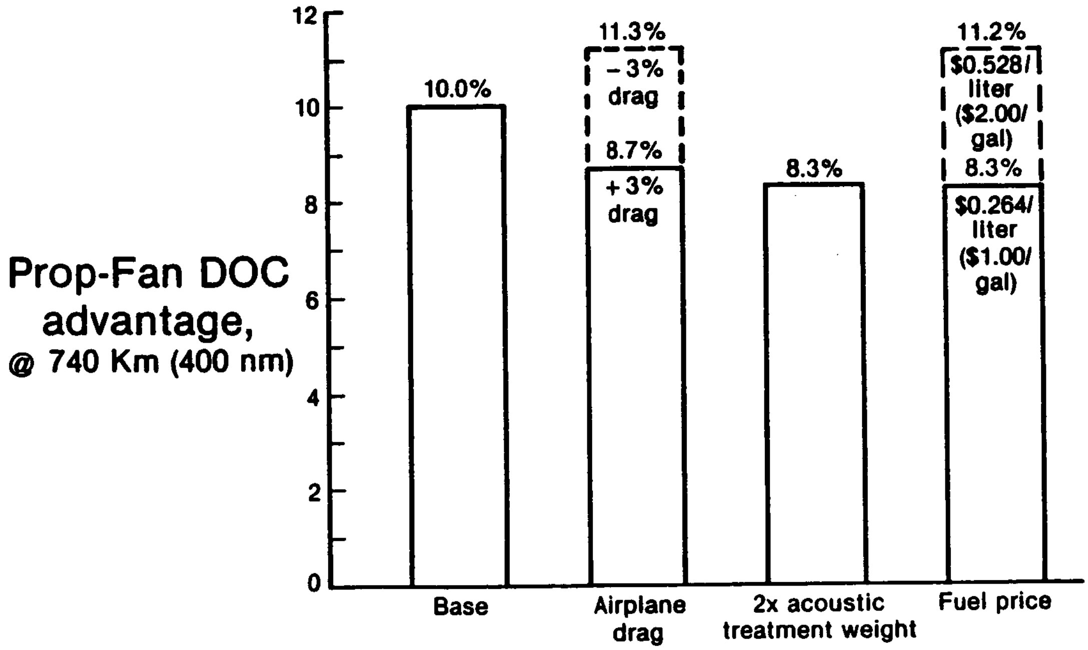

Figure 4.4-14 Effect of Key Parameters on the Direct Operating Cost Advantage of the Prop-Fan Powered Aircraft - The direct operating cost advantage of the Prop-Fan powered aircraft is not significantly reduced by increased airplane drag or acoustic treatment weight, or by lower fuel prices. (J27638-82)

The first bar in each figure shows the (fuel burn/DOC) advantage demonstrated by the Prop-Fan powered aircraft sized for 10,668 m (35,000 ft) over the turbofan under the basic ground rules of the APET Program. The next bar shows the impact of a 3% drag penalty on the Prop-Fan powered aircraft. (Recall that in the basic ground rules, no interference drag penalty was imposed on the Prop-Fan powered aircraft relative to the reference turbofan.) Also shown is the potential benefit which can be derived by the Prop-Fan powered aircraft from recovering part of the propeller slipstream residual swirl. More extensive wind tunnel testing of Prop-Fans installed on wings will be required to accurately assess the benefit, or penalty, of the interaction of the propeller slipstream and the wing. The third bar shows the effect of doubling the fuselage acoustic treatment weight in the Prop-Fan powered aircraft. Under the basic ground rules, the Prop-Fan powered airplane had 907 kg (2000 lb) more fuselage treatment than the turbofan powered airplane. (Acoustic treatment has a greater effect on direct operating cost than engine weight because it includes the associated airframe cost increase while engine weight and cost are treated separately.) The fourth bar, seen only in Figure 4.4-14, shows the effect of changing fuel price from the baseline of $0.396/liter ($1.50/gallon) to $0.264/liter ($1.00/gallon) and $0.528/liter ($2.00/gallon). All of these effects are shown individually, not cumulatively.

The effects of airplane drag and acoustic treatment weight may seem to disagree somewhat with the Prop-Fan influence coefficients shown previously. This is due primarily to a shift in the base. The influence coefficients were based on the performance of the Prop-Fan powered aircraft, while the effects of increased airplane drag and acoustic treatment weight are measured in terms of the advantage of the Prop-Fan powered aircraft over the turbofan powered aircraft. Hence, these effects are based on the performance of the turbofan powered airplane. Since the turbofan powered aircraft has higher fuel burn and direct operating costs, changes in the performance of the Prop-Fan powered aircraft appear smaller when compared to the performance of the turbofan powered airplane. For example, a one percent change in the fuel burn of the Prop-Fan powered aircraft would produce a 0.8% change in the advantage of the Prop-Fan powered aircraft over the turbofan powered airplane.

Figures 4.4-13 and 4.4-14 indicate that the performance advantage of the Prop-Fan powered aircraft is large enough to withstand significantly greater penalties in interference drag and acoustic treatment weight than have been assumed for the APET Program. In addition, the direct operating cost advantage of the Prop-Fan powered aircraft remains large (over 8%) at fuel prices equivalent to 1981 levels.

4.4.7 Acoustics

The flyover noise of the Prop-Fan powered airplane was estimated at the certification points defined by the FAA Part 36 Chapter 3 regulations. The predicted Effective Perceived Noise Level (EPNL) was established by summing the noise generated by each of the noise sources during the airplane flyover. The procedure, described in Appendix C of Reference 1, is comprised of a Hamilton Standard supplied module to predict the noise generated by the Prop-Fan and an engine noise procedure, developed by Pratt & Whitney, based on turbofan engine noise data. The results of the flyover noise estimates are presented in Figure 4.4-15 and indicate that the Prop-Fan powered airplane will meet the noise rules by a comfortable margin.

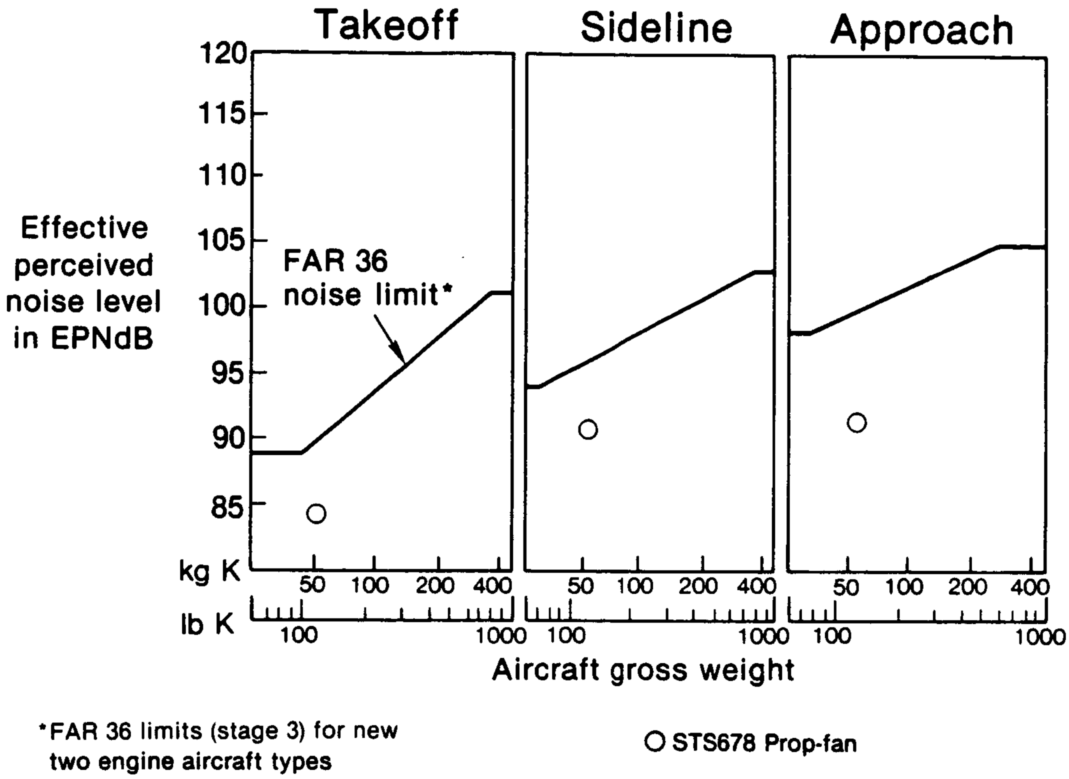

Figure 4.4-15 Flyover Noise Predictions for the Prop-Fan Powered Airplane - The Prop-Fan powered airplane will satisfy FAR Part 36 Stage 3 noise regulations by a comfortable margin. (J27638-148)

The contribution of each of the component noise sources to the total airplane noise is tabulated in Figure 4.4-16. The major noise sources in the Prop-Fan powered aircraft are the Prop-Fan (propeller) at the takeoff and sideline conditions and the Prop-Fan and airframe at approach.

	EFFECTIVE PERCEIVED NOISE LEVEL (EPNL)		
	FAA Noise Point		
Component	Takeoff	Sideline	Approach
Prop-Fan	83.1	90.2	82.1
Airframe	67.7	69.1	88.9
Combustor	57.8	59.9	65.4
Jet	67.6	69.2	51.4
Compressor	32.8	34.2	75.2
Turbine	32.1	31.2	59.2
Gearbox	34.1	35.0	38.3
Total	84.4	90.9	91.7

Figure 4.4-16 Prop-Fan Airplane Component Noise Levels - The major noise sources are the Prop-Fan at takeoff and sideline conditions and the Prop-Fan and airframe at approach. (J27638-913)

The Prop-Fan is quieter than conventionally designed propellers at the same tip speed and loading due to the low noise features developed over the last several years by the NASA-Lewis Research Center and the Hamilton Standard Division of United Technologies Corporation (References 2 and 3). At the maximum power condition (sideline and takeoff) the other noise sources are all at least 10dB below the noise of the Prop-Fan. At the low thrust approach condition, the Prop-Fan noise is reduced appreciably because of the low disk loading and lower tip speed, and airframe noise predominates. It is noted that the airframe noise prediction is based on a correlation of data obtained from turbofan powered airplanes and, therefore, does not include the possible effect of noise generated by the prop wash impinging on the wing surface. This possible noise source should be evaluated as part of a large scale flight test program.

The flyover noise of the turbofan powered airplane was also predicted by the procedure described in Reference 1, with many of the engine component modules common to both predictions. In order for the turbofan powered airplane to meet the FAA noise regulations, it was necessary to provide two noise reduction features. First, turbomachinery noise was attenuated through the use of acoustic lining in the inlet, fan case, and fan and turbine exhaust ducts. The amounts of lining used in each section were based on Pratt & Whitney full scale engine acoustic liner experience. Based on this experience, 2.04 m^2 (22 ft^2) of liner material was apportioned to the inlet, 5.67 m^2 (61 ft^2) to the fan case and fan duct, and 1.18 m^2 (12.7 ft^2) to the turbine exhaust case. The inlet, fan case and fan duct lining material design involves a wire mesh bonded to perforated plate with a honeycomb backing. The acoustic lining was predicted to provide acceptable noise levels at the sideline and approach conditions. Second, a two-slope takeoff (reduced thrust "cutback") procedure was used in order to meet the FAA rules for that community noise point. The predicted levels for the turbofan powered aircraft relative to the Prop-Fan powered aircraft and for both aircraft relative to the FAA rules for two-engine airplanes are shown in Figure 4.4-17. The altitudes at takeoff and sideline for the two aircraft are shown in Table 4.4-VIII. At the sideline condition where the respective altitudes differ by only 60 m (200 ft), the noise levels of the Prop-Fan and turbofan powered airplanes are very close. At the takeoff condition, the Prop-Fan powered aircraft attains a much greater altitude than the turbofan powered aircraft; 865 m (2840 ft) vs 600 m (1970 ft). The greater altitude provides larger attenuations of the Prop-Fan noise relative to the sideline condition than those associated with the smaller altitude change and reduced thrust of the turbofan, with the result that the Prop-Fan powered aircraft is quieter at takeoff. At approach, the predictions indicate that the fan is the dominant source of noise in the turbofan, whereas airframe noise dominates in the Prop-Fan powered aircraft. As a result, the noise level of the Prop-Fan powered aircraft is less than the noise level of the turbofan powered aircraft.

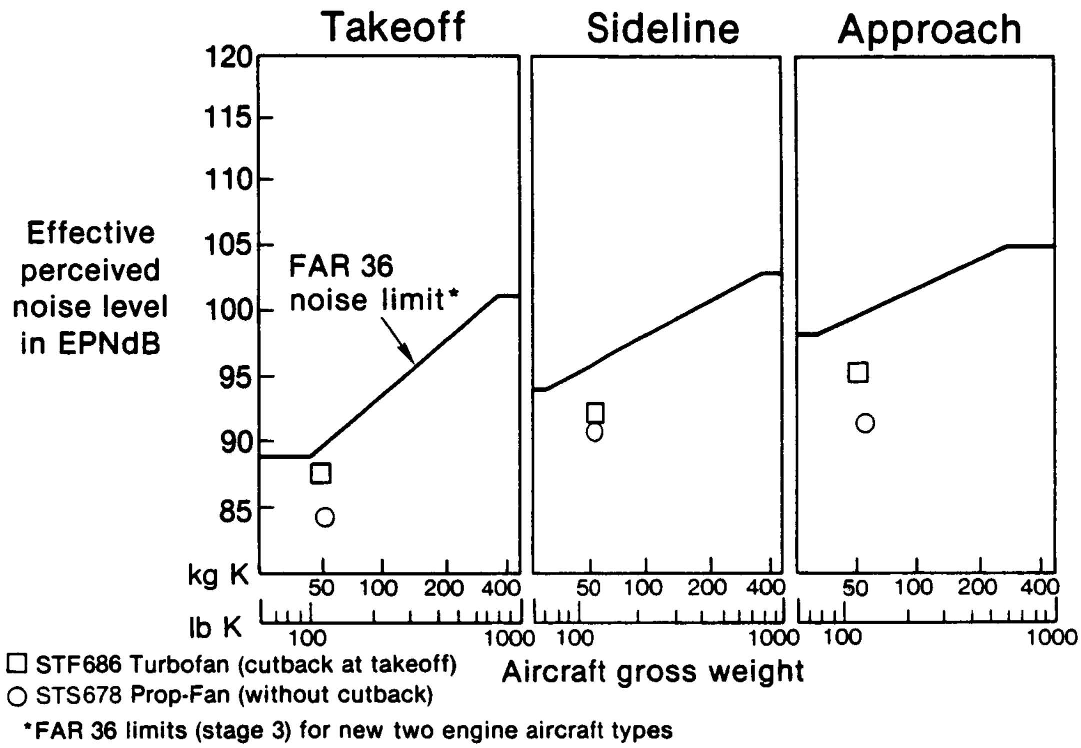

Figure 4.4-17 Comparison of Flyover Noise Levels for Prop-Fan and Turbofan Powered Aircraft - The overall noise level of the Prop-Fan powered aircraft is lower than the noise level of the turbofan powered aircraft. (J27638-149)

TABLE 4.4-VIII
ALTITUDE AT CERTIFICATION FLYOVER LOCATIONS

	Takeoff	Sideline	Approach
Prop-Fan Powered Aircraft - m (ft)	865 (2840)	426 (1400)	120 (394)
Turbofan Powered Aircraft - m (ft)	600 (1970)	365 (1200)	120 (394)

It should be noted that the predictions for the Prop-Fan powered aircraft are based on a method for Prop-Fans which has not been verified with full scale data. Further, the turboshaft engine horsepower is well beyond the range of existing turboshaft engines, with components that are closer in design philosophy to current turbofan engines than to current turboprops. Thus, even though the current prediction methodology indicates that the Prop-Fan powered airplane is an attractive concept in terms of community noise considerations, comfortably meeting the FAA noise rules, full scale verification is required.

4.4.8 Emissions

The emissions goals of the International Civil Aviation Organization were used in the APET Program. These goals, presented in Table 4.4-IX, are referred to as "Research Goals" for newly certified engines. The advanced Mark V combustion system which is projected to be available for 1992 engine certification will provide the capability to meet these emissions goals.

TABLE 4.4-IX
INTERNATIONAL CIVIL AVIATION ORGANIZATION
Emissions Research Goals

	Research Goals (g/kN)*	
	Turbofan	Prop-Fan
Unburned Hydrocarbons	4.35	4.35
Carbon Monoxide	42.0	42.0
Oxides of Nitrogen	56.6	54.0
Smoke (SAE Number)	24.7	24.4

* Thrust at Sea Level Takeoff Static Conditions in kilonewtons

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SECTION 4.5 -- DISCUSSION OF RESULTS
Task V -- Advanced Prop-Fan Engine Technology Plan

4.5 TASK V - ADVANCED PROP-FAN ENGINE TECHNOLOGY PLAN

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4.5 TASK V - ADVANCED PROP-FAN ENGINE TECHNOLOGY PLAN

4.5.1 Introduction

The objectives of Task V were: (1) to identify the key technology components for an advanced Prop-Fan engine system assuming technology verification by 1988 for in-service use in the early 1990's and (2) to prepare a key technology development and verification plan to the subcomponent design, fabrication, and test level with appropriate schedules and costs.

In response to these objectives, Pratt & Whitney has identified three key technology areas unique to the advanced Prop-Fan engine system: (1) large-size reduction gear, (2) Prop-Fan/nacelle/inlet/compressor interactions, and (3) small-size high-pressure compressor technology. Pratt & Whitney has prepared detailed technology verification plans for these key areas including schedules and estimated costs for planning purposes. Cost data for these technology verification plans will be submitted under separate cover.

In addition, Pratt & Whitney has identified further study effort required in the area of engine/aircraft integration. This effort includes, in order of descending priority:

- Free vs non-free power turbine engine
- Propulsion system mounting
- Engine/aircraft heat rejection
- Integrated propulsion system/aircraft control

These studies must be performed jointly between engine and airframe manufacturers and may lead to identification of other key technology areas.

The technology plan and the definition of the study requirements are based not only on the technical work performed as part of this contract, but draw on technical work and planning done by Pratt & Whitney as a subcontractor to Hamilton Standard under the NASA-sponsored Counter Rotation Contract (NAS3-23043). The study has also benefitted from the extensive work on the Prop-Fan system that has been conducted by Pratt & Whitney since 1980 using company funds.

To ensure Prop-Fan propulsion system technology verification by 1988, the current study contract must be followed by initiation of the large-size reduction gearbox preliminary design in 1983 and the start of NASA-sponsored test programs to evaluate Prop-Fan/nacelle/inlet interactions no later than 1985.

In Section 4.5.2, background and requirements for the three technology verification programs are discussed briefly and the overall plan is presented. Detailed discussions of each program are presented in Sections 4.5.3 through 4.5.5. Engine/aircraft integration studies, which were started in this program and should be continued, are discussed in Section 4.5.6.

4.5.2 Key Technology Components and Overall Verification Plan

Pratt & Whitney has identified three key technology areas that are unique to Prop-Fan propulsion systems and which cannot be addressed by either current turboprop or turbofan engine experience. These technology issues must be verified in a timely manner to provide the technology base required before engine manufacturers can commit to a full-scale advanced turboprop engine development program leading to certification. These key technologies are:

- Large-size reduction gear
- Prop-Fan/nacelle/inlet/compressor interactions
- Small-size high-pressure compressor technology

Large-Size Reduction Gear

There is no experience in the Western world with advanced turboprop reduction gears in the 10,000 to 15,000 shaft horsepower (shp) class. Experience to date is based on small 5000 shp class units with initial designs dating back some 25 years. Therefore, a modern technology base must be established for the higher horsepower ratings.

In addition, airline operators of the current 5000 shp reduction gears have experienced less than satisfactory durability and high maintenance costs. Improved technology with respect to bearings and seals, gears, materials, structures and lubricants will significantly improve reliability and efficiency resulting in lower cost and reduced weight. To launch a successful Prop-Fan powered aircraft, the airline industry must be convinced that the reliability and maintainability of the reduction gear will match that of components in current turbofan engines.

The pitch change mechanism, which is an integral part of a gearbox design, can have a significant adverse impact on overall gearbox reliability and maintenance cost. Therefore, it is important that the advanced turboprop gearbox technology program include pitch change control requirements and considerations.

Prop-Fan/Nacelle/Inlet/Compressor Interactions

The higher flight speed of a turboprop aircraft with Prop-Fan propulsion presents an all new Prop-Fan/nacelle/engine inlet environment significantly more severe than the environment in today's engines. At cruise operation, the inlet face Mach number will be near Mach 1 with blade passing pulsations superimposed. Consequently, current design methodology and test verification techniques must be extended to produce an efficient installation.

Design and test verification is required to determine the interactive performance of the Prop-Fan, nacelle, and inlet with the engine. First, the flow field behind the Prop-Fan must be defined, including the interactive effects of the inlet and the nacelle. This will define the spillage and flow distortions at the inlet front face and will provide input to the inlet design.

Second, inlet designs have to be tested to evaluate pressure recovery and flow stability of the boundary layer. Third, the interactive effects of the Prop-Fan and the inlet with the compressor (engine) will have to be determined to assure that the combined performance of the inlet and the compressor (engine) is stable and efficient.

Small-Size High-Pressure Compressor Technologies

Engines of the future can be expected to have higher overall pressure ratios than engines currently in service. This will result in smaller blade heights in the rear stages of the high-pressure compressor. These small blades are susceptible to erosion and early loss of performance. This condition is aggravated in advanced turboprop engines because of the smaller size core compared to turbofan engines of equal thrust. Centrifugal compressors can provide relief for this problem. However, the application of a centrifugal compressor at the rear of a high pressure ratio compression system will require new structural and aerodynamic technology.

Studies conducted in Task II of this program highlighted the favorable acquisition cost and maintenance cost characteristics of a turboprop engine using centrifugal or mixed-flow compression in the high spool. (A mixed-flow compressor stage is similar to a centrifugal compressor stage except that the flow at the exit has an axial velocity component, in addition to a large radial component.) A systematic analytical/test program is required to determine the relative cost/benefit ratio of using all-axial rear stages vs centrifugal or mixed-flow rear stages in advanced engines. In addition, a technology program is required to establish fabrication processes and computer codes for design of advanced centrifugal or mixed-flow compressors.

Overall Program Plan

The overall advanced Prop-Fan engine technology plan has been formulated to attain verification of key technologies by 1988 with engine certification in 1992. The major elements of this six-year plan are shown in Figure 4.5-1 and summarized below:

- Large-size reduction gearbox program culminating in the test of a full-scale gearbox in a back-to-back rig.

- Prop-Fan/nacelle/inlet/compressor aerodynamic interaction program consisting of: (a) evaluation of the Prop-Fan/nacelle/inlet interactions, (b) inlet/diffuser testing, (c) analytical code development, and (d) evaluation of the compatibility of the best inlet with the compressor (the effects of the Prop-Fan and nacelle will be simulated at the inlet throat).

- Small-size high-pressure compressor technology program.

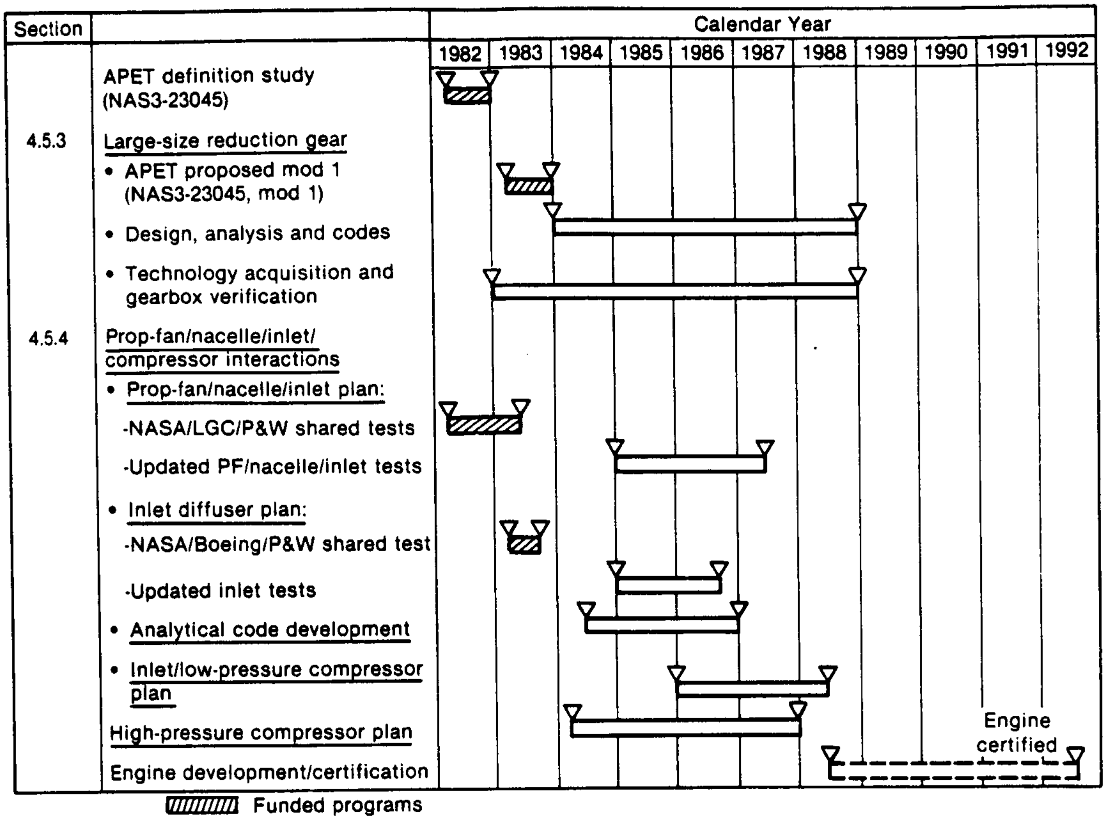


Figure 4.5-1 Overall Advanced Prop-Fan Engine Technology Program Plan - This program, which was initiated with the APET Study contract in 1982, provides the capability to certify an advanced turboprop engine for a Prop-Fan powered aircraft in 1992. (J27638-200)

The successful accomplishment of this program, along with NASA's ongoing Large-Scale Advanced Prop-Fan (LAP) program and the flight test of the propeller as part of the Propeller Test Assembly (PTA) program, will provide the technology data base necessary for industry to commit substantial development funds to certification of an advanced turboprop aircraft in 1992.

The technical effort in each phase of the program is discussed in detail in the following sections.

4.5.3 Large-Size Reduction Gearbox and Pitch Control Plan

Current experience with turboprop reduction gearboxes is based on small 5000 shp class units with initial designs dating back some 25 years. Airline operators complained about the less than satisfactory durability, low mean time between removals (MTBR), and high maintenance cost associated with these designs.

New design methods and technologies for bearings and seals, gears, materials, structures and lubricants must be developed for advanced turboprop reduction gears in the 10,000 to 15,000 shaft horsepower class. These improved capabilities will be used to produce a reduction gearbox with significantly greater reliability and efficiency, lower cost and reduced weight, and a Prop-Fan pitch control that is effectively integrated with the reduction gear system. The ability of this design to meet commercial performance and service goals must be verified by large-scale rig tests before industry will initiate development of a commercial Prop-Fan propulsion system.

The gearbox design objectives discussed in Task III are summarized below. The overall design goals address the major concerns of current operators of turboprop engines.

Overall design goals

- Reliability (MTBR) - Hours 15,000
- Cruise Efficiency - Percent 99.0

Improved modularity and reduced maintenance costs

- Externally mounted aircraft accessories
- Accessible oil pump and condition monitoring systems
- Propeller brake pad
- "On-the-wing" main shaft seal replacement
- Accessible propeller pitch control components

4.5.3.1 Objectives and Benefits

The overall objective of the Large-Size Reduction Gearbox/Pitch Control Technology and Research Program and the expected benefits are summarized in Table 4.5-I.

TABLE 4.5-I
OBJECTIVES AND BENEFITS
Large-Size Reduction Gearbox/Pitch Control Program

Objective: Verify the efficiency and durability of a large-size reduction gearbox/pitch control system using 1988 technology.

Benefits:

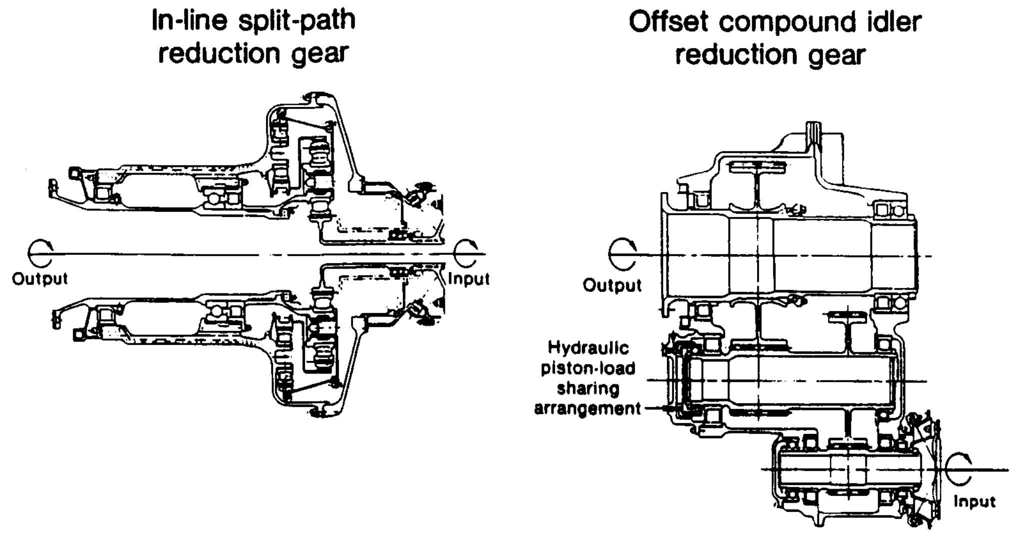
o Reduce the gearbox heat rejection by 50% at cruise (i.e., improve efficiency from 98% to 99%).

o Provide the potential to reduce oil cooler size and drag 40% - 60%.

o Improve gearbox/pitch control durability (MTBR) 100% to 200%.

o Reduce weight 10% to 20%.

In Task III of the current contract, the best in-line and offset reduction gearbox concepts were selected for further study. These concepts are illustrated in Figure 4.5-2. The major features of each system are summarized in the figure.



- o Lowest Weight
- o Smallest Diameter

- o Best Efficiency
- o Lowest Cost
- o Most Reliable
- o Lowest Maintenance Costs

Figure 4.5-2 In-Line and Offset Flight Gearbox Designs - Both reduction gear configurations were selected as promising candidates for a Prop-Fan propulsion system. (J27638-199)

Integrating the Prop-Fan pitch control with the gearbox is a particularly important consideration with an in-line gearbox system. This topic was discussed in some detail in Sections 4.3.1.2.2 and 4.3.1.7. With the in-line gearbox configuration, mechanical and hydraulic hardware for the Prop-Fan pitch change control must pass through the gearbox. Studies to reduce the complexity of the gearbox/pitch control system and improve modularity are included in the preliminary design efforts discussed below.

4.5.3.2 Gearbox/Pitch Control Technology and Research Plan

The Large-Size Reduction Gearbox/Pitch Control Program Plan is illustrated in Figure 4.5-3. The plan includes: Design and Analysis, Component Technology Acquisition using new and existing test rigs, and Large-Size Gearbox/Pitch Control Verification.

The three phases in the program are discussed below.

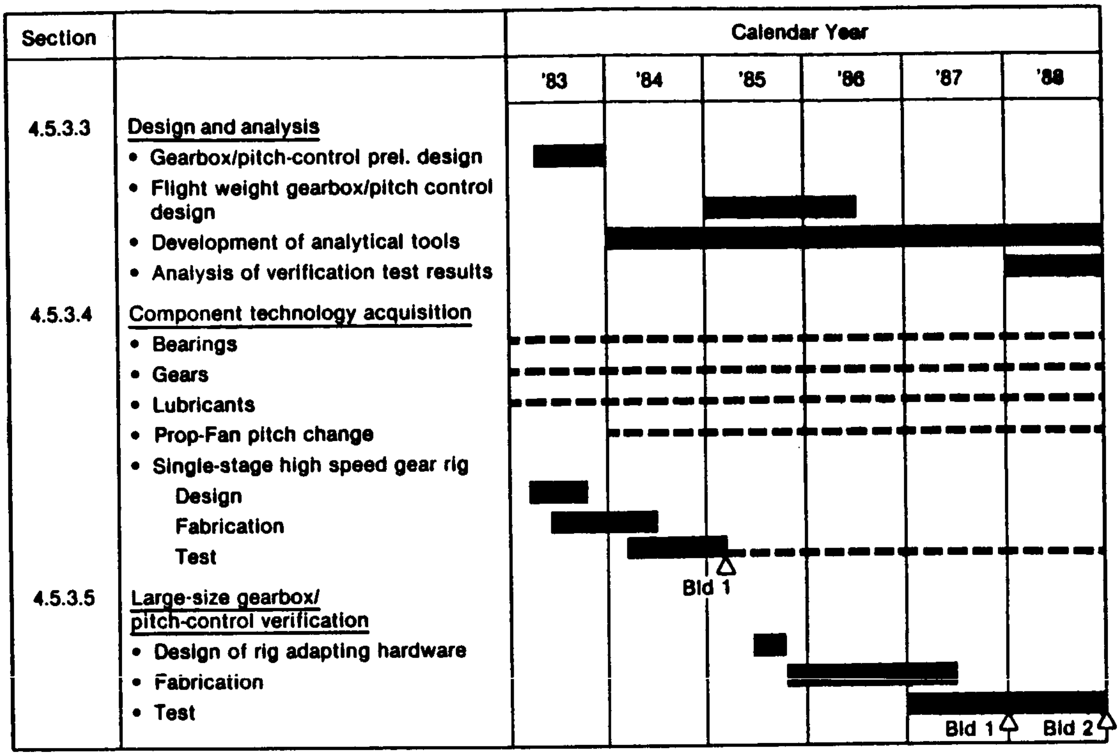


Figure 4.5-3 Large-Size Reduction Gear Technology Program Plan - This three-part effort covers design and analysis, technology acquisition, and technology verification efforts. (J27638-188)

4.5.3.3 Design and Analysis

This phase will lead to the design of a viable reduction gearbox in the 10,000 to 15,000 shp class as well as development of analytical tools which can be used to evaluate and refine the design. There are three segments in the design and analysis phase: (1) a preliminary design effort to quantify the benefits of advanced technology and to confirm the advantages of modular maintenance concepts, (2) design of a flight weight (10,000 - 15,000 shp class) gearbox/pitch control, and (3) development of analytical design tools using data acquired from the subsequent component technology acquisition and large-size gearbox verification test efforts. A brief description of each of these efforts follows.

Gearbox/Pitch Control Preliminary Design

During the APET contract effort, two reduction gear systems were selected for a Prop-Fan propulsion system: an offset compound idler gearbox and an in-line split path planetary gearbox. These gearbox concepts are illustrated in Figure 4.5-2. With NASA approval, a preliminary design of one of the two gearbox/pitch control concepts, capable of being demonstrated in a large size by the late 1980s, will be compared with a state of the art (1983) gearbox design to assess the benefits of advanced (1988) technology. The design of the associated pitch control mechanism will include an assessment of completely new or enhanced conventional designs which will improve modularity and reduce maintenance costs. At the end of this phase full-scale preliminary design drawings of the gearbox/pitch control system will be generated. Partial detailed and assembly drawings will be produced to support the maintenance, modularity and mounting assumptions. The most critical technical consideration is whether in-line gearbox/pitch control integration and modularity can be improved significantly. The planned analytical effort will address this issue.

Flight Weight Gearbox/Pitch Control Design

The design phase has been structured to take advantage of evolving technologies within the APET program as well as related developments in the field. Initiating the design phase in 1985 ensures that the latest benefits of the component technology acquisition effort, which covers bearings, gears, lubricants, and the pitch change mechanism, will be incorporated in the flight weight gearbox. Related Government and industry programs will be reviewed periodically.

Development of Analytical Design Tools

The development of analytical design tools is an extremely important counterpart to the preliminary design and flight weight gearbox design efforts. Computer codes used in gear, bearing, housing and lubrication system design will be updated as new test data becomes available. The major areas of emphasis are listed in Table 4.5-II. Information would be obtained from the APET Component Technology Acquisition and Large-Size Gearbox Verification programs, as well as other related NASA, Government and industry sponsored efforts.

TABLE 4.5-II
COMPUTER CODE DEVELOPMENT WILL BE ADVANCED IN CRITICAL AREAS

o Efficiency	- Replace classical gear friction factors with elastohydrodynamics (EHD) based theory - Verify bearing loss in "SHABERTH"
o Durability	- Calibrate bearing derived analysis with gear pitting data - Derive EHD related scoring limits
o Dynamics	- Demonstrate Hamilton Standard's gear system dynamics code
o Cooling	- Develop planetary system oil jet design methods based on high velocity, multiple jet concepts
o Condition Monitoring	- Develop failure mode/progression rate analytical models

The analytical design improvements specified in Table 4.5-II will fill voids in the current design data base, validate new methods, and extend current methods to new regimes as required.

To meet advanced performance goals, precise power loss prediction methods are required. Gear loss analysis can be enhanced by recent advances in lubrication theory in which lubricant film thickness and sheared film heat generation are calculated using elastohydrodynamic concepts. Specific bearing loss calculations will be developed for the single row spherical roller bearing considered optimal for high speed planetary gears. Appropriate gear and bearing thermal models will be assembled in "SHABERTH" or an equivalent computer program and calibrated against data from individual bearing, gear and transmission test rigs.

Durability testing of gears and bearings will be conducted in various rigs. All test results will be used to calibrate and enhance existing prediction methods. An area of particular interest is the effect of gear speed and oil jet velocity and flow rate on the scoring probability of a high speed planetary gear set, where heating of sun gear teeth is particularly intense.

Dynamic tooth loads, tooth profile correction and system stiffness and damping requirements are particularly critical in a planetary gear system where torsional and radial ring modes may combine to cause fatigue or wear related failures. The Hamilton Standard computer models for gear system dynamics can be compared with data from the multi-purpose gear rig over a wide range of operating conditions.

New technology is needed to minimize gearbox losses without compromising durability. New oil delivery concepts for planetary gears and bearings and new analytical methods to optimize these concepts will aid in meeting this objective.

The need for development of highly effective condition monitoring data acquisition and processing systems will be supported with suitable testing in both the technology acquisition and gearbox verification phases. Reducing the results of these tests to an appropriate mathematical model of key failure modes and failure progression rates will ensure that maintenance actions can be made cost effective without compromising flight safety.

Analysis of Verification Tests

Analytical design procedures developed during the program will be used to predict large-size gearbox performance. These predictions will then be compared with data from the gearbox verification tests. This final comparison of predicted and demonstrated results will aid significantly in evaluating gearbox technology levels and provide a well-established data base for future verification efforts.

4.5.3.4 Component Technology Acquisition Phase

The component technology acquisition phase consists of two segments: (1) testing of individual component technology acquisition rigs, and (2) testing of the advanced technology components in a single-stage rig to verify the efficiency and durability of the individual components in an integrated system. Current plans call for adapting individual component rigs available at NASA and in industry to test advanced technology concepts. An advanced technology, back-to-back, single-stage, high-speed gear test rig will be used to assess a variety of aspects of reduction gear technology in an integrated gear system.

Individual Component Rigs

The APET technology acquisition program will take advantage of existing gearbox test rigs used at NASA, Government agencies and other divisions of United Technologies Corporation. The rigs described below are suitable for "generic" technology programs which would benefit a wide range of gearbox applications.

Component technology acquisition rig testing would focus on several key areas: (1) heat generation and dynamic gear tooth load scoring limits, (2) static tests to measure ultimate strength of single gear teeth, (3) one way bending fatigue gear tooth testing, (4) reverse bending fatigue gear tooth testing, and (5) roller bearing testing to assess thermal performance and durability characteristics which cannot be established in a large-size back-to-back gearbox test rig in a cost effective manner. Single bearing, single mesh, and tooth bending fatigue test rigs are illustrated in Figures 4.5-4, 4.5-5 and 4.5-6 respectively.

Bearings - The single bearing test rig shown in Figure 4.5-4 will provide essential data on planet bearing friction and wear, as well as lubrication and cooling. It will also be used to evaluate the rolling contact fatigue properties of candidate gear materials for a pinion bearing with an integral gear and bearing outer ring, and to evaluate the effects of outer ring thickness on bearing performance.

Gears - The single mesh gear rig shown in Figure 4.5-5 will be used to evaluate gear tooth form and to develop materials and lubricants. Among the tooth forms considered for evaluation are the high contact ratio buttress form and the noninvolute constant relative radius of curvature form which can reduce scoring tendencies produced by high sliding velocity.

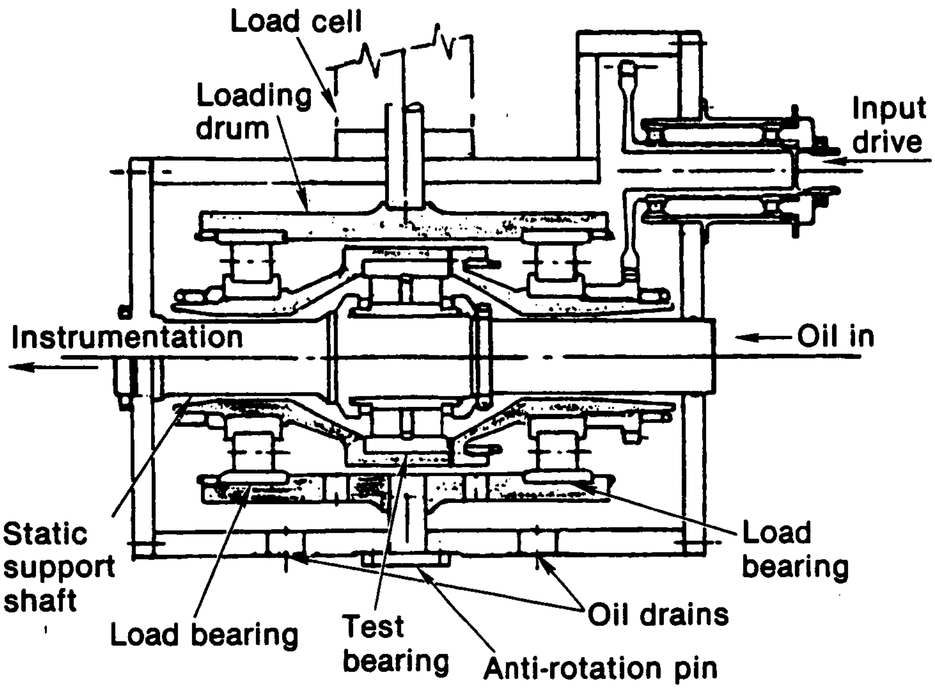


Figure 4.5-4 Single Bearing Test Rig - This rig will provide data on planet bearing friction and wear. (J27638-125)

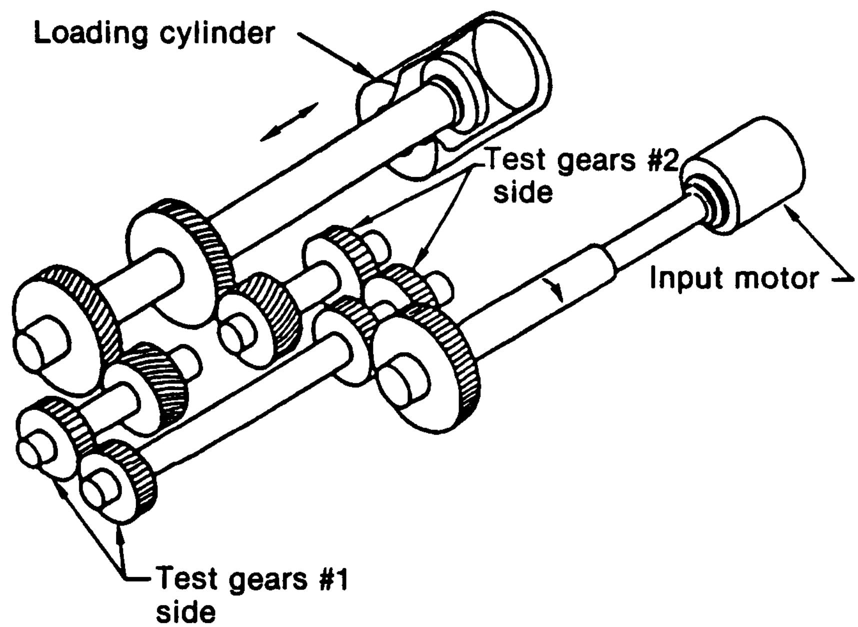


Figure 4.5-5 Single Mesh Gear Test Rig - This rig will be used to evaluate gear tooth forms, and to evaluate materials and lubricants. (J27638-131)

The gear materials tests would compare scoring and pitting resistance of materials such as Vasco X-2 and Carpenter EX00053, the major candidates to replace AIS19310. Processing variables would also be examined. Lubricant tests would compare promising new oils with MIL-L-23699.

The tooth bending fatigue rig shown in Figure 4.5-6 would be used to compare the bending strengths of candidate materials and root geometry, as well as to evaluate processing options such as the unground fillet produced by the Maag tooth grinding machine.

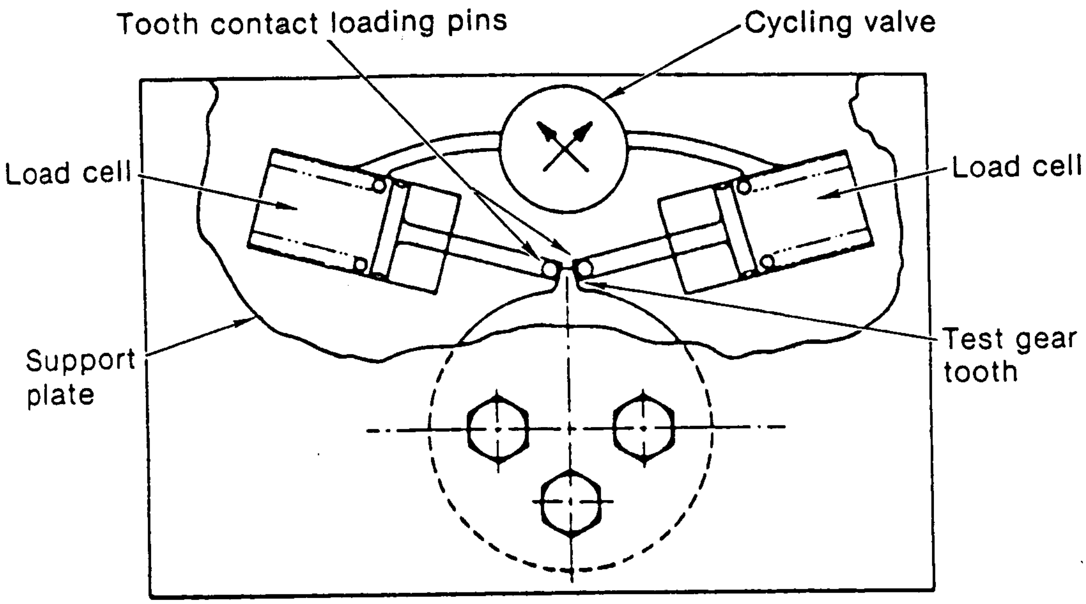


Figure 4.5-6 Gear Tooth Bending Fatigue Rig - This rig will be used to evaluate the bending strength of gear tooth materials. (J27638-126)

Lubricants - Lubricant technology will also be acquired using these rigs. Increases in oil inlet temperature, allowable temperature rise, oil load carrying ability and flash temperature index will be assessed. It is expected that Synthesized Hydrocarbon Fluids (SHF) will be used in this program.

Prop-Fan Pitch Change - Current pitch change technology is based on proven concepts involving hydraulic actuators and mechanical signal transmission. Advanced pitch change systems may incorporate new concepts such as digital fiber optic control and rare earth electric drives. To implement these advanced systems, in-depth study will be required in the following areas:

o Define a safety philosophy. For example, three signal channels would be required for a fail-operational system, while two channels would be required for a fail-fixed system.

- Define a diagnostic system compatible with the safety philosophy.
- Improve optic encoder technology for measuring blade angle and phase.
- Develop reliability and maintenance values for fiber optic components.
- Develop methods for transferring fiber optic signals across the rotating boundary of the Prop-Fan .
- Develop methods for electric power generation and control in the propeller.

During the preliminary design effort for the gearbox/pitch control, it is likely that specific technology needs will be identified. These pitch change technology considerations will form a basis for future technology verification plans and component acquisition programs.

Single-Stage, High-Speed Gear Rig (Multi-Purpose Gear Rig)

A new single-stage gear technology acquisition rig will be designed to provide generic gear, bearing and lubricant technology applicable to future single rotation or counter rotation Prop-Fans. It will address the following advanced technology issues: (1) high speed integral gear(s) and spherical roller bearings, (2) high capacity gear tooth form including high contact ratio, (3) modulating and/or high velocity lubrication and cooling system technology, (4) condition monitoring systems to identify techniques that would lead to improved durability, (5) advanced materials for gears, bearings, and housings, (6) advanced lubricants, (7) technologies for controlling noise and/or vibration, and (8) potential pitch control integration issues. The generic technology information obtained with this rig will be equally applicable to both offset and in-line gearboxes.

The single stage feature of the rig provides room for maximum instrumentation. Thus, significantly more technical data will be obtained from the gear set than in a standard back-to-back rig.

The rig will be designed to test a planetary gear set at any combination of speed, power, torque, and temperature as well as with a variety of lubrication systems appropriate for the Prop-Fan application.

The center portion of the rig contains the hydraulic thrust layshaft torque loader system. This system includes the torque reaction spur gear mesh and the torque application helical gear mesh. This portion of the rig also includes the speed control for the ring gear, as well as temperature and stress instrumentation for the components being tested. The speed reduction test gear set and a similar speed increasing gear set are located at each end of the test rig. This arrangement isolates the gear sets from each other and from the torque load application system in the center section of the rig. Isolation of the gear sets also assures precise control over operating conditions and dependable test data. The test section will be thoroughly instrumented with strain, temperature, and displacement devices required for the sun gear, ring gear, and planet carrier components. It may be possible to obtain some planet gear data using optical devices such as infrared temperature sensors. Direct access will be provided to test hardware to ensure rapid inspection and modification of components.

The complete test installation shown in Figure 4.5-7 includes the multi-purpose gear rig and an electric motor used to drive the planet carrier of the multi-purpose gear rig. The design of the rig ensures easy access to instrumentation and the torque loader hydraulic connections. A bed plate has been used to ensure alignment between the electric motor drive and multi-purpose gear rig. Two ring gear drive systems ensure proper speed and power transfer to the ring gear in the multi-purpose rig.

The installation has been designed to permit test rig attitude to be varied, allowing lubrication systems to be evaluated at simulated flight conditions.

Rig Design - The multi-purpose gear rig incorporates the design flexibility to test single-stage gears for a variety of applications, as well as to accommodate back-to-back tests of a complete gear system for the turboprop engine. However, much more instrumentation can be used in single-stage tests. Ample radial and axial clearances are provided to ensure that current slip rings and telemetry instrumentation hardware can be used for data acquisition. Sufficient insulation is provided around the test vehicle to guarantee accurate temperature measurements which are used to determine the efficiency of the gear meshes.

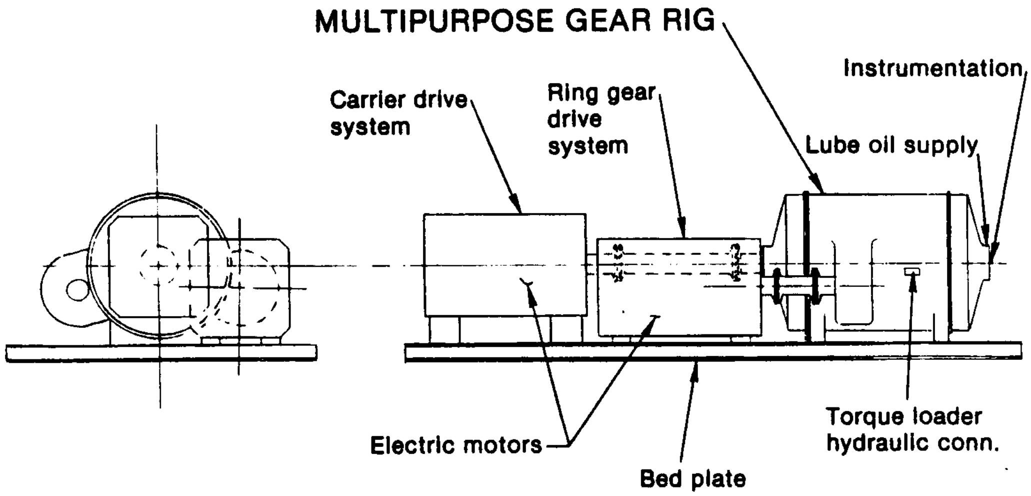


Figure 4.5-7 Test Rig Installation - The major components in the gearbox test rig are shown in this figure. (J27638-205)

Rig Fabrication - The rig is fabricated in two major sections: (1) the basic core or center portion of the gearbox rig, and (2) the gearing and shafts for the single-stage planetary gear set. Rig construction includes all cases, covers, housings and supports for the static parts. Rotating parts include shafts, gears, gear shafts, and telemetry slip ring adapting equipment.

Test Program - Test plans for the single-stage planetary gear rig call for a wide range of operating conditions to be applied to each build. The program will provide as much fundamental data as possible on gear system performance, dynamics, lubrication and cooling, and condition monitoring operating constraints in order to fully calibrate the analytical design systems.

Lubricant supply and scavenge concepts will be critical to high efficiency and optimal cooling effectiveness; therefore, several versions will be examined early in the test schedule. When proper oil system function and target efficiency levels have been obtained, the rig can be used for durability and condition monitoring studies.

Rig tests will include verification of selected materials and lubricants, as well as demonstration of condition monitoring concepts.

4.5.3.5 Large-Size Gearbox Technology Verification

The single-stage rig provides much needed generic technology for the design of the large-size gearbox. However the large-size gearbox verification test is required to establish the confidence of the airline industry in a large-size gearbox with modern technology

The large-size gearbox technology verification program consists of three phases: (1) design of rig adapting hardware required to test the flight weight gearbox/pitch control in the multi-purpose rig, (2) fabrication of the flight weight gearbox/pitch control system and rig modifications, and (3) verification testing of the gearbox. The schedule for these efforts is shown in Figure 4.5-3.

Design of Rig Adapting Hardware

The large-size gearbox technology verification rig uses the center portion of the multi-purpose gear rig. Adapting hardware will be designed for the gearbox verification test to accommodate the unique instrumentation and hardware at both ends of the rig.

Fabrication

This effort covers fabrication of rig adapting hardware and fabrication of the flight weight gearbox/pitch control system. (The design of this system was discussed in Section 4.5.3.3.) The rig adapting hardware includes cases, covers, housing and supports for static structures, as well as shafts, gears, gear shafts, gear sets, bearings, and minor parts for the rotating structures. The bed plate, drive motors, etc. for the major facility have already been completed. However, to ensure maximum use of time, a second basic core will be fabricated for the multi-purpose gearbox rig. This will allow large-size gearbox verification hardware to be assembled while the individual component technology acquisition tests are being conducted.

Test Program

Two tests (builds) of the large-size gearbox rig are planned. The first build of the gearbox will incorporate the latest gearbox technology available at the time of design. The second build (mid-1988) will incorporate technologies acquired in the individual component and single-stage technology acquisition rigs. The schedule shown in Figure 4.5-3 will permit more than three years of technology acquisition to be incorporated in the second build. Test results will be compared with the analytical design data base, as indicated previously.

4.5.4 Prop-Fan/Nacelle/Inlet/Compressor Plan

It is expected that the higher flight speed and unique Prop-Fan configuration of an advanced turboprop system will create an all-new Prop-Fan/nacelle/ inlet/ compressor environment significantly worse than the environment in today's engines.

Before a Prop-Fan powered aircraft can be designed with confidence, the critical questions of the aerodynamic interaction between the propeller, nacelle, and inlet must be addressed. Historically, inlets for turboshaft engines have been designed for substantially lower flight speeds than those being considered for the Prop-Fan. Hence, design techniques were used which are totally unsuited to the higher speed Prop-Fan applications. For example, the aerodynamic characteristics of conventional turboshaft engines are: (1) cruise speeds at Mach numbers around 0.4, (2) flow diffusion up the spinner to Mach numbers of approximately 0.2, (3) flow captured by the inlet at such low speeds that the diffusion process is minimal and can be performed in almost any manner without incurring high losses. With the Prop-Fan, however, the aerodynamic designer is confronted with the following challenges: (1) cruise Mach numbers up to 0.8 introducing a potential for high drag, (2) flow diffusion up the spinner, but reaccelerating to transonic Mach numbers immediately behind the propeller, (3) high velocity flow which must be captured by the inlet and scroll diffused around a large-size gearbox down to velocities low enough to enter the compressor while still keeping the losses and distortions to acceptable levels, and (4) flow distortions at the compressor front face that are expected to be significantly worse than the flow distortions in today's engines. The compressor must be stable under these distortions and in addition attenuate them to a level that the rear compressor can handle.

4.5.4.1 Objectives and Benefits

The objectives and benefits of the proposed Prop-Fan/nacelle/inlet/compressor interactive program are summarized in Table 4.5-III.

4.5.4.2 Prop-Fan/Nacelle/Inlet/Compressor Plan

Ideally, tests to check out these potential problems would be conducted with the real inlet shape and with the nacelle and wing simulated. This is not possible with existing Prop-Fan test rigs because of mechanical constraints. For instance, the shafting required to drive the two-foot diameter scale model propeller is too large to allow simulation of the inlet internal flowpath. Similarly, evaluating inlet flow in an airplane model requires duct areas too large for a simulated pylon.

Therefore, to address the interactive concerns, a building block approach was structured using different test rigs. Figure 4.5-8 shows the four elements that will be used to address the aerodynamic concerns mentioned previously. Figure 4.5-8a illustrates a Prop-Fan rig with an aspirated inlet. Figure 4.5-8b is a schematic of a large-scale inlet diffuser test rig. This type of testing can provide important external aerodynamic data, inlet front face distortion, propeller performance and propeller stress during both forward flight and reverse.

TABLE 4.5-III
OBJECTIVES AND BENEFITS
Prop-Fan/Nacelle/Inlet/Compressor Plan

Objectives

- Determine the flow field behind the Prop-Fan, including the effects of the inlet and the nacelle.
- Determine inlet contours that will minimize spillage loss and distortions at the front face, maximize pressure recovery, and attenuate distortions through the inlet contours to the compressor front face.
- Determine the impact of the Prop-Fan exit flow and the inlet contours on low-pressure compressor performance, stability and attenuation characteristics.
- Develop analytical techniques to enable future three-dimensional, non-axisymmetric, transonic flow design capability.

Benefits

- Prop-Fan/nacelle/inlet flow field interactions defined.
- Inlet pressure recovery and distortion attenuation defined as a function of inlet contours.
- Inlet/low-pressure compressor interactions defined with the effects of the Prop-Fan and nacelle included.
- Inlet/low-pressure compressor stability limits and high-pressure compressor inlet conditions defined.
- New analytical techniques developed and verified for future Prop-Fan applications.

The rig program shown in Figure 4.5-8c will be a full scale inlet/compressor test to verify stability behind a Prop-Fan/inlet. The distortions measured at the inlet throat during the Prop-Fan/inlet interaction tests will be simulated with screens in the inlet/compressor rig. Downstream of these screens, the inlet and drive shaft aerodynamic configuration will be modeled exactly. Inlet throat and compressor face distortion, surge margin loss, blade stress, and compressor efficiency data will be recorded during this test.

Figure 4.5-8d depicts a half-airplane model which will be tested with powered nacelles (props) to provide propeller/wing interaction data. The airframe manufacturers, in concert with NASA-Langley, are taking the lead in this area. However, the design of the propulsion system is heavily impacted by these external aerodynamic concerns, so Pratt & Whitney will continue to closely monitor progress.

The proposed schedule for these rig programs, excluding the half-airplane model program, is shown in Figure 4.5-9. A schedule for the analytical effort supporting the inlet rig programs is also shown. Each major phase of the program is described in greater detail below.

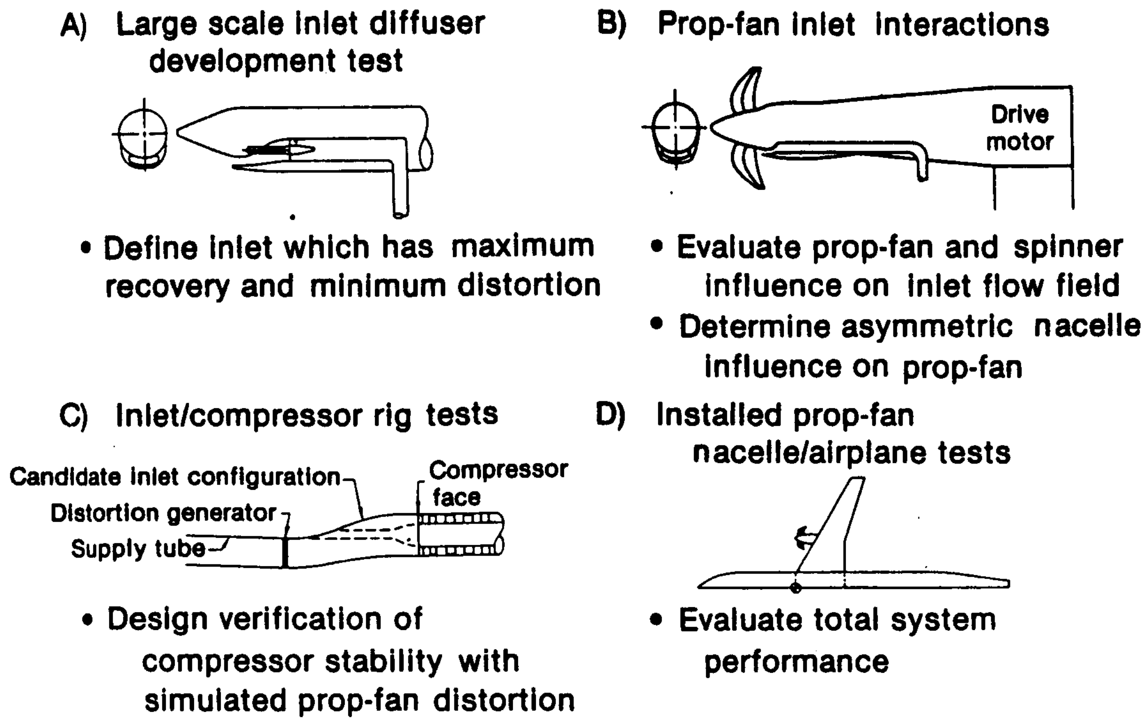


Figure 4.5-8 Building Block Technology Plan - This approach permits a comprehensive evaluation of Prop-Fan/nacelle/inlet/compressor interactions using current test rig technology. (J27638-189)

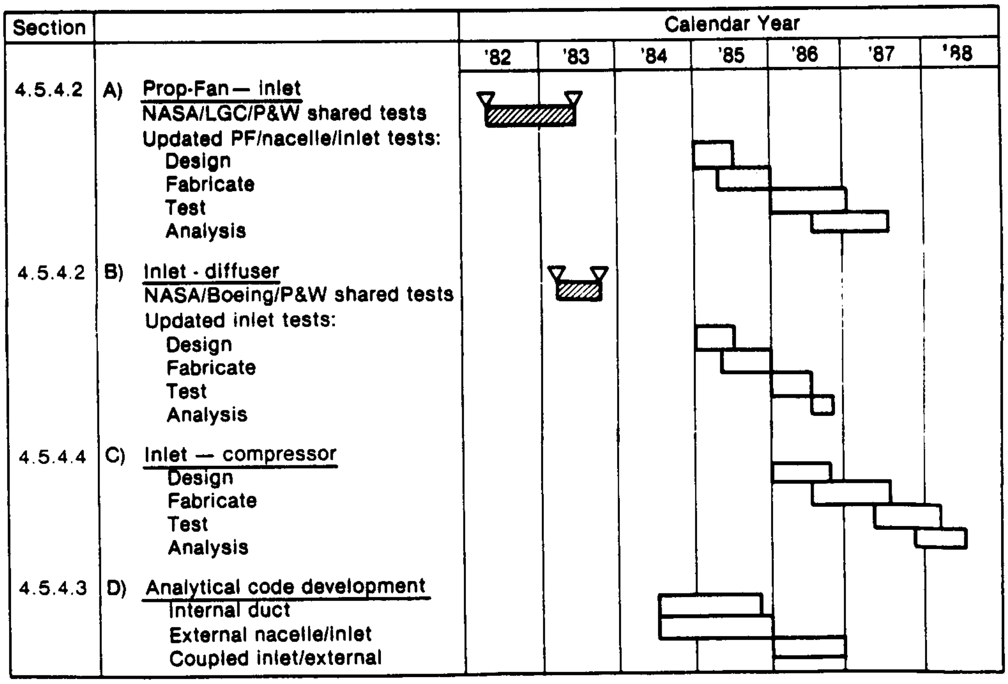


Figure 4.5-9 Prop-Fan/Nacelle/Inlet/Compressor Interaction Program Schedule - The schedule includes three rig test programs and development of new analytical code. (J27638-190)

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Prop-Fan/Nacelle/Inlet Plan

Previous Prop-Fan scale tests conducted by NASA have shown transonic Mach numbers near the inner diameter wall immediately behind the propeller and in the area where the inlet would be located.

In the NASA model, shown in Figure 4.5-10, an axisymmetric body was located behind the propeller. It is known that the presence of a flowing inlet will change the local aerodynamic characteristics, but it is not known to what extent. This local change introduces additional concerns, including blade stress, vibration, and performance losses. For these reasons, Pratt & Whitney, Lockheed-Georgia Corporation and Hamilton Standard are collaborating with NASA in an experimental program to obtain data on Prop-Fan/nacelle/inlet interactions. In this program, an aspiration system was designed, fabricated, and mated to the United Technologies Research Center (UTRC) propeller test rig to allow simulation of flowing inlets. A program plan was formulated, model hardware was fabricated and instrumented, and testing was initiated in November, 1982 using the SR3 propeller provided by NASA-Lewis. Testing is continuing and results are being analyzed. These shared tests provide a good base from which to proceed to the accompanying test plan.

However, this testing has not addressed some of the important technological unknowns: (1) nacelle drag has not been measured, (2) parametric variations of inlet aspect ratio, inlet proximity to the Prop-Fan, and boundary layer diversion also need study, and (3) the inlet types (chin, bifurcated, trifurcated, and annular) all must be tested to obtain data required for trade studies and configuration selection. These tests should be performed with the most promising propellers over a range of speeds, with variable blade angles at the flight conditions of interest.

Figure 4.5-10 Prop-Fan Model Test Rig - This two-foot diameter model is being used in ongoing NASA Prop-Fan tests. (82-A-9026-001)

An additional area of concern is operation during reverse. Depending on the method of reversing (propeller rotation through a flat pitch vs feather), it is possible that either in reverse or during transition, the inlet would be "starved" for airflow or might ingest highly distorted wakes. Data are needed to evaluate the seriousness of this potential threat as well.

There are two facilities available for conducting these tests: the NASA-Lewis wind tunnel and the wind tunnel at the United Technologies Research Center (UTRC). It appears that the NASA-Lewis propeller test rig is dedicated to the demonstrator program. Since the aspiration system is available on the UTRC drive rig, this seems to be the likely place to perform the tests. However, a skin balance will have to be designed, fabricated and calibrated to measure nacelle drag. With the success UTRC has enjoyed in recent propeller testing, the development of the skin balance appears to be the only remaining technical challenge in the execution of this program.

The test program outlined in Figure 4.5-11 will provide key external aerodynamic information to resolve the interactive effects of the Prop-Fan/nacelle/inlet.

A description of the design, fabrication, test and analysis portions of this plan follows.

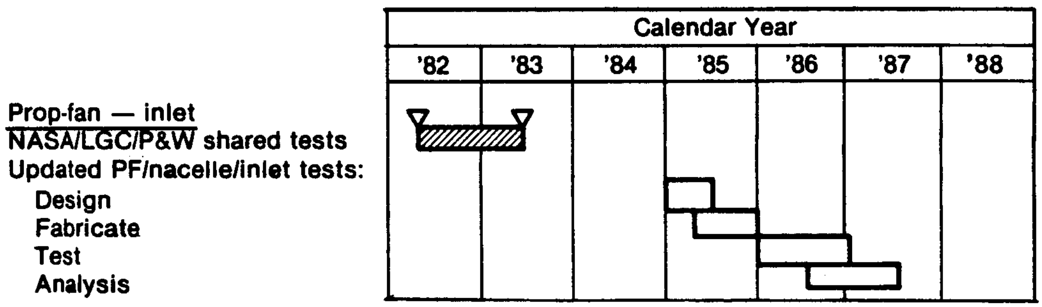

Figure 4.5-11 Prop-Fan/Nacelle/Inlet Plan - This program will provide key external aerodynamic information to resolve the interactive effects of the Prop-Fan, nacelle, and inlet. (J27638-914)

Design - A comprehensive test plan will be developed for the wind tunnel program. Test configurations that provide the most meaningful information on Prop-Fan/nacelle/inlet interactions will be selected, and detailed models of the configurations will be designed. The design process will include aerodynamic flowpath definition as well as the actual mechanical design. Flowpath definition will be based on advanced three-dimensional and transonic inlet analysis codes, ensuring that the test configurations will be separation-free, with minimum shock losses. A qualified vendor will be selected to complete the mechanical design and fabricate the models. Pratt & Whitney will closely monitor production to ensure that program schedules are met and wind tunnel safety requirements satisfied.

Fabrication - Test models will be fabricated and instrumented by a qualified vendor, using existing propellers, hubs, and drive rigs. The construction of the models should be straightforward. However, it is recommended that additional hardware be fabricated to improve the productivity of the test program:

- An additional set of SR3 propeller blades and hubs should be manufactured to permit simultaneous testing in the NASA-Lewis and UTRC facilities.
- Multi-component balances should be designed, fabricated, calibrated and developed.
- An aspiration system should be constructed for the NASA-Lewis wind tunnel.

The aspiration system allows inlet airflow to be simulated at low speed, high angle of attack operation. The system also permits meaningful drag data to be obtained with the flowthrough nacelles used in high speed tests.

Test - Two types of tests will be conducted in the wind tunnel facilities: (1) low speed, high angle of attack tests simulating takeoff operation, conducted primarily to evaluate inlet distortion, and (2) high speed tests focusing on inlet distortion and external drag. The key variables in the test program are listed below:

- Tunnel Mach number
  - Low speed; 0 to 0.25
  - High speed; 0.6 to 0.8
- Propeller blade angle and rpm
- Inlet mass ratio (0.6 to 1.0)
- Inlet configuration

Chin, bifurcated and annular inlet configurations will be evaluated in the program. To ensure a comprehensive evaluation of the inlets, the following parameters should be varied systematically:

- $D_{highlight}/D_{throat}$ to determine distortion
- Inlet throat orientation to evaluate swirl
- Throat area and aspect ratio to permit trade studies of drag vs pressure recovery
- Boundary layer diverter

Data Analysis - Measurements of throat total pressure distributions will be used to calculate throat recovery and back pressure effects as a function of propeller advance ratio, propeller power and inlet mass flow ratio (see Figure 4.5-12). Drag data will be acquired using skin balance measurements. It will then be used to optimize boundary layer diverter geometry (see Figure 4.5-13). Drag data will also be used to perform trade studies of drag vs inlet recovery, ensuring optimum specific fuel consumption (see Figure 4.5-14).

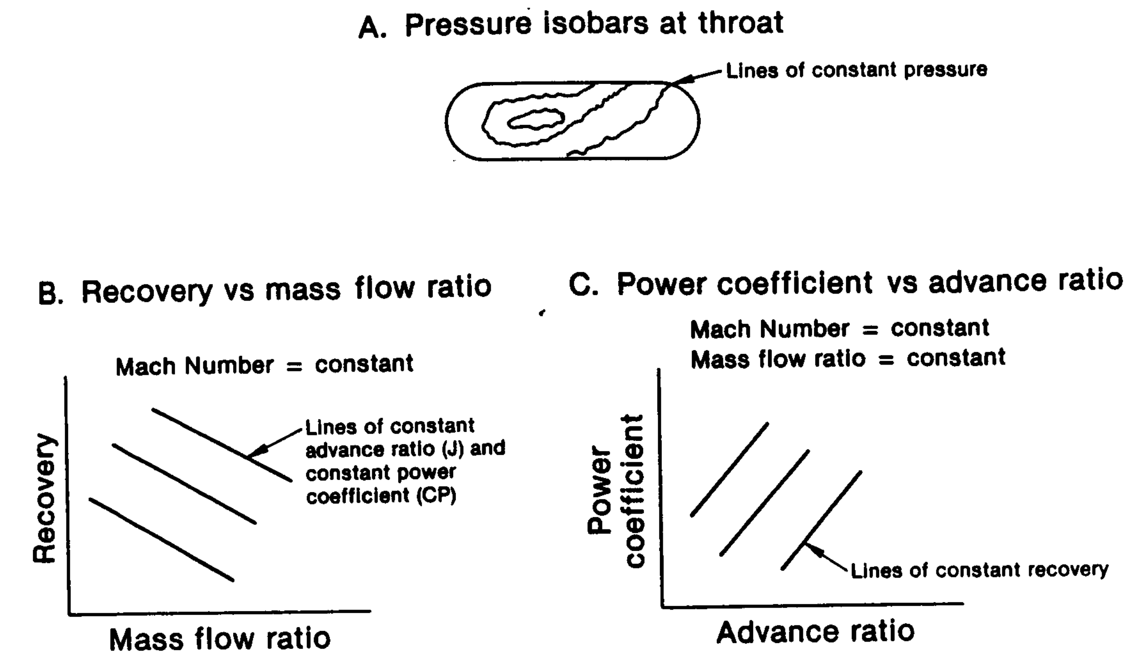

Figure 4.5-12 Inlet Throat Recovery - Throat pressure distributions will be used to compute throat recovery and propeller back pressure effects. (J27638-236)

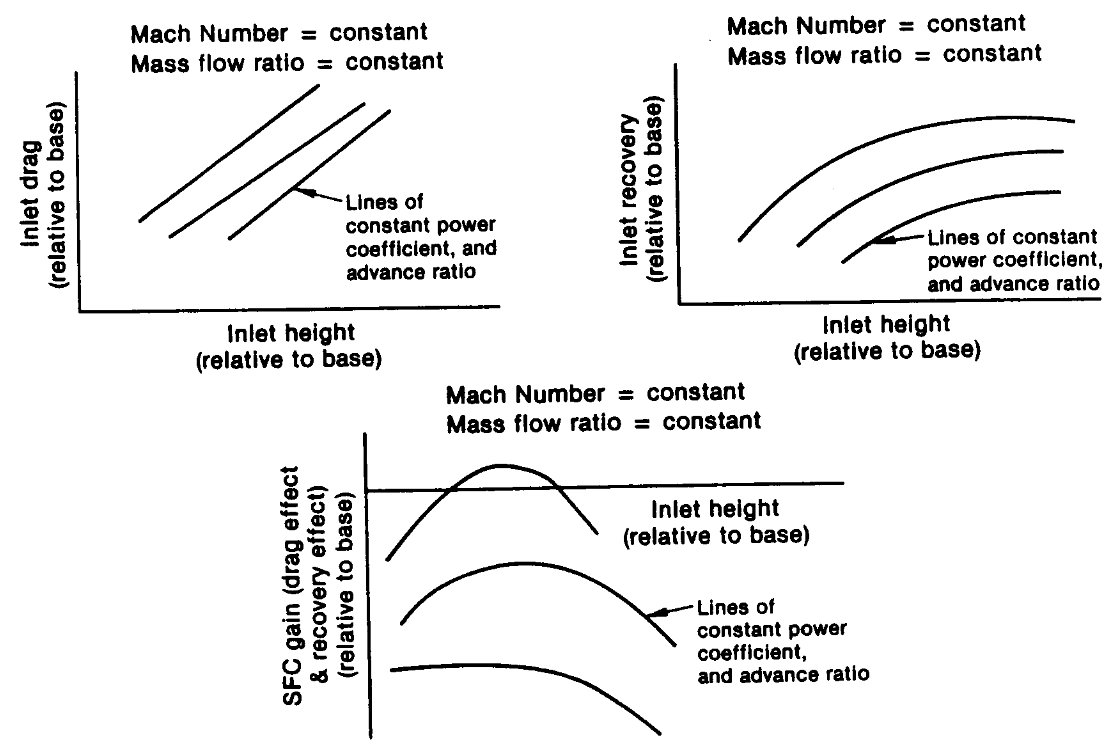

Figure 4.5-13 Optimization of Boundary Layer Diverter Height - Drag data will be used to optimize boundary layer diverter geometry. (J27638-207)

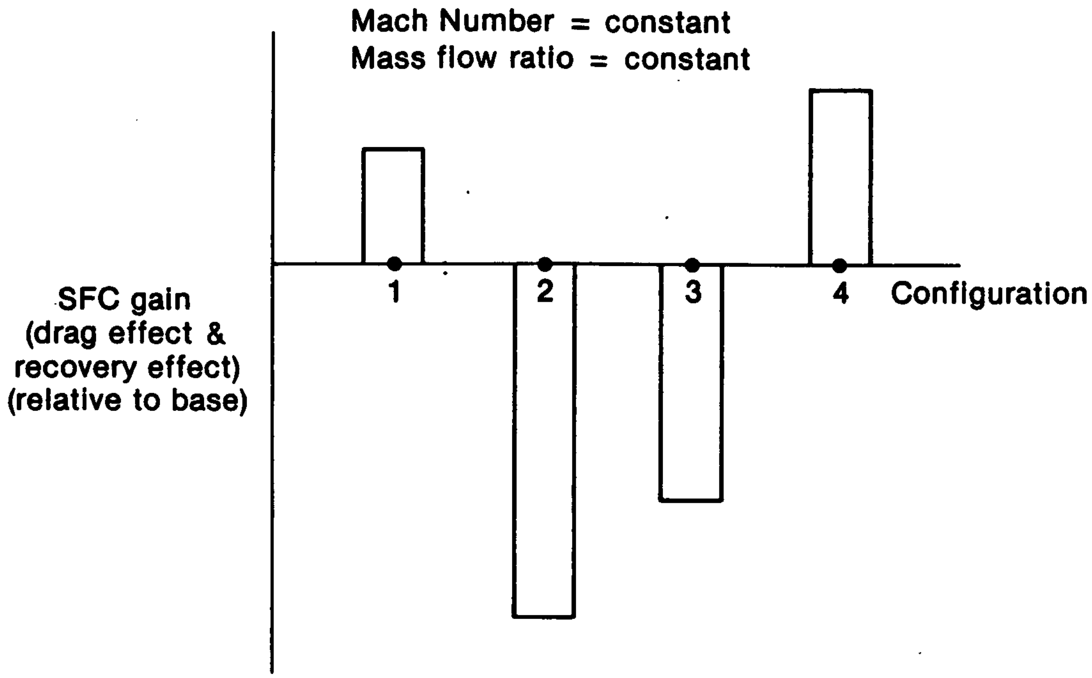

Figure 4.5-14 Optimization of Configuration at Best Inlet Height - Selection of the optimum inlet height will minimize fuel consumption. (J27638-208)

Inlet/Diffuser Plan

The selection of the best inlet for the Prop-Fan application involves trades between several key factors, including weight, distortion, pressure recovery and spillage drag. In addition, the type of gearbox configuration, in-line or offset, can be a significant factor in the inlet selection process. APET study results indicate that either chin or bifurcated inlets are preferable for an offset gearbox, while bifurcated or trifurcated inlets would be preferred with an in-line gearbox.

To illustrate, changing the inlet configuration from chin to bifurcated to trifurcated to annular constitutes a transition in which more and more of the propeller hub circumference is used by the inlet (see Figure 4.5-15). As this transition occurs, less circumferential scrolling is required and the inlet diffuser can be shortened. A shorter diffuser in turn allows the shaft from the compressor to the gearbox to be shortened. Other favorable results of using more of the circumference are a reduction in inlet height, which lowers the threat of bird strikes, and a reduction in localized propeller back-pressurization. However, increasing the number of inlets also produces some negative results, such as more difficult boundary layer diversion and an increase in the wetted area of the inlet diffuser, which reduces the pressure recovery. Obviously, detailed trade studies are required to arrive at the "best" inlet for any given application. The Inlet/Diffuser Program plan shown in Figure 4.5-16 will provide important internal aerodynamic information as a function of the key inlet design variables.

A description of the design, fabrication, test and analysis phases of the program follows.

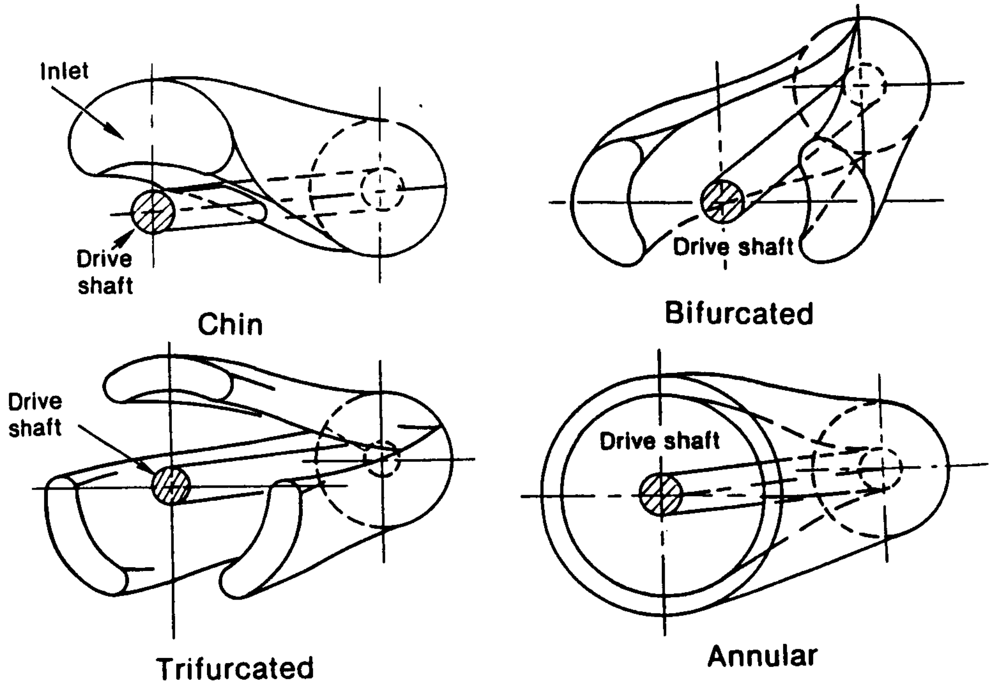

Figure 4.5-15 Candidate Inlets - The best inlet for a Prop-Fan propulsion system will depend on the specific application. (J27638-104)

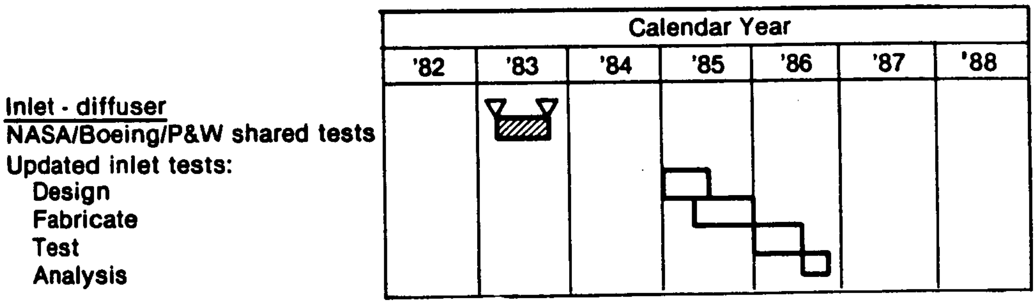

Figure 4.5-16 Inlet/Diffuser Program Plan - This program will provide critical internal aerodynamic information. (J27638-915)

Design

A comprehensive test plan will be developed for the Inlet/Diffuser Program. After the test configurations have been selected, agrodynamic flowpaths will be defined using advanced computer codes; detailed models of the configurations will be designed and fabricated by a qualified vendor.

In the flowpath definition, an appropriate flow code will be used to determine the three-dimensional pressure distributions on the wall of the inlet. The pressure distributions are input to a boundary layer analysis to check for flow separation. Since a three-dimensional boundary layer analysis is not yet available, individual segments, or "strips," of the duct will be analyzed. However, this type of analysis introduces a significant degree of uncertainty, a deficiency which constitutes one of the major reasons for the inlet diffuser program.

To reduce length and weight, the contours in the model should be approaching the very edge of flow separation. The model contours will then bracket the predicted onset of separation; some of the contours will be designed conservatively, with gentle flow turning, while others will incorporate aggressive diffusion and flow turning.

Once the flowpath has been defined, the mechanical design and fabrication of the models will be completed by a qualified vendor. The inlet/diffuser rig in which the models will be tested is illustrated in Figure 4.5-17.

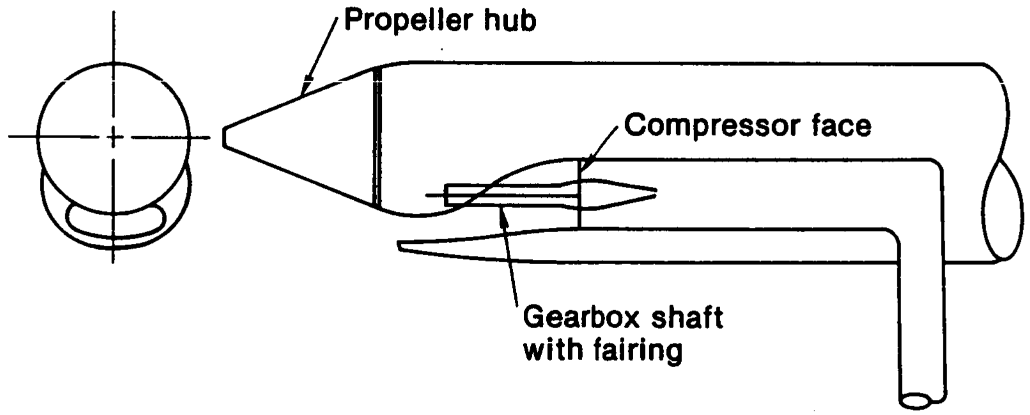

Figure 4.5-17 Inlet/Diffuser Development Rig - The large scale inlet/diffuser test will provide empirical information on the inlet internal aerodynamics including the effects of propeller hub, gearbox shafting and compressor face boundary conditions. (J27638-211)

Fabrication - The models will be made of fiberglass and will be constructed in halves to permit flow visualization. Advanced computer-aided design and computer-aided manufacturing techniques, including numerical control machining, will be used to minimize costs and expedite production.

Test - A large scale inlet diffuser development test is recommended to obtain empirical information on the internal aerodynamics of the inlet. There are two major reasons for large scale testing: (1) the aerodynamic threat to the diffuser is flow separation, which is sensitive to Reynolds number, and (2) with high aspect ratios, the inlet height becomes very small, making it difficult to instrument the model without creating blockage problems.

Current plans call for the tests to be performed in a low speed wind tunnel with an aspiration system using a bladeless hub/inlet combination. Many of the test configurations being considered for the Prop-Fan/nacelle/inlet interaction tests are being designed to reduce total pressure distortion across the inlet. If these designs are successful, the diffuser development testing can be conducted with a bladeless spinner/inlet combination in a low speed tunnel. If the designs are not successful the model will have to be more sophisticated. It should incorporate screens of varying solidity to obtain a representative inlet throat total pressure profile, as well as turning vanes to simulate swirl.

Chin, bifurcated and annular inlet configurations will be evaluated in the program (as indicated in Figure 4.5-15). The annular configuration, which is most amenable to analytical treatment, will require minimal test evaluation.

Instead, the tests will focus on chin and bifurcated inlets to validate the analytical codes for these more complicated configurations. The major test variables include:

Throat to compressor face area ratio
Area distribution through the diffuser
Maximum wall curvature
Length
Compressor face (simulated) Mach number
Lip radius
Aspect ratio
Effectiveness of vortex generators
Angle of attack

Compressor face total pressure patterns, wall static pressure distributions, and flow visualizations will be recorded. These data will be used to evaluate inlet pressure recovery and map regions of flow separation.

The tests should be conducted over a range of Mach numbers at inlet flows covering idle through maximum climb. Either the NASA-Lewis 8-foot by 6-foot wind tunnel or the main wind tunnel at the United Technologies Research Center could be used for the tests. Both facilities are of sufficient size and have aspiration systems which can accommodate models as large as half-scale.

Data Analysis - The inlet/diffuser tests will provide internal inlet performance data (no propeller) for half-scale candidate inlets. Compressor face total pressure distributions will be used to calculate compressor face recovery and compressor face distortion (see Figure 4.5-18). Static pressure distributions and oil flow techniques will be used to define lip and diffuser separation (see Figure 4.5-19).

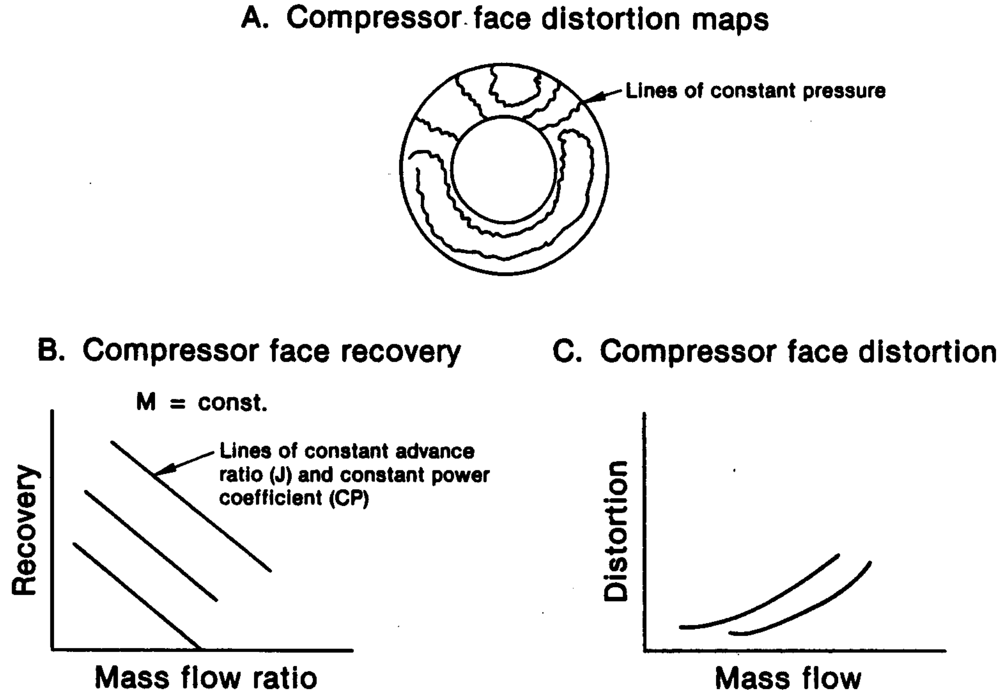

Figure 4.5-18 Compressor Face Conditions - Total pressure distributions will be used to compute compressor face recovery and compressor face distortion. (J27638-209)

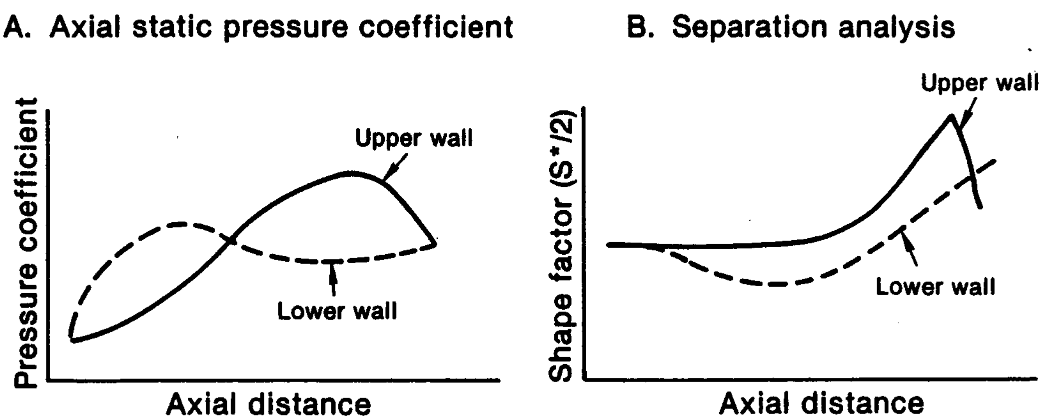

Figure 4.5-19 Separation Analysis - Static pressure data and oil flow techniques will be used to define lip and diffuser separation. (J27638-210)

4.5.4.3 Analytical Code Development

To generate high performance designs for Prop-Fan/nacelle/inlet/compression systems, rig test programs must be complemented by development of analytical codes that will permit refinement and extrapolation of the experimental data. Relying exclusively on empirical correlations requires extensive refinement of the configurations through repeated wind tunnel tests. This type of approach is very costly. Further, there are several inherent shortcomings in wind tunnel testing: (1) model mounting support systems and wall interference effects cast doubt on the accuracy of wind tunnel results at transonic speeds; (2) the limited Reynolds number capability of existing wind tunnel facilities introduces large risks in extrapolating wind tunnel results to full-scale conditions; and (3) current instrumentation and flow visualization techniques provide only limited knowledge of the flow field around the aircraft. It is becoming clear that these concerns can be mitigated when applicable computational tools are available. Computational aerodynamic methods have been used for a wide range of design problems in turbofan applications. These codes are flexible, and with some modification can be adapted to the turboprop application.

However, the problem of adequately modeling the aerodynamics of Prop-Fan/nacelle/inlet/compressor interactions is complicated by the presence of strong viscous effects, arising from shocks, premature separation, etc. An analysis of a steady state problem must be quite general; therefore, the governing equation system will be highly nonlinear. Typically, solutions for these systems are only possible with the simplest of geometries, such as flat plates, not for the complex propeller/nacelle/inlet geometry. Figure 4.5-20 lists several codes that can be adapted to the Prop-Fan system. The upper portion of the figure shows an inlet internal duct analysis which would model the inlet/diffuser results. The center portion of the figure identifies codes which, at the present time, seem most applicable to the external flow field analysis representing Prop-Fan/nacelle/inlet testing. All of these codes need some refinement, adaptation, and calibration before they can be used with confidence on the highly three-dimensional Prop-Fan inlet problem. Ultimately, the appropriate external analysis will be coupled with the internal analysis. The figure shows a schedule for completing the modifications to the codes. The following paragraphs describe the codes and the refinements required.

Two approaches to internal flow analysis were considered: (1) a potential solver in which paneling methods are used to describe the flowpath, and (2) PEPSI(G) a fully viscous forward marching Navier-Stokes solver. Pratt & Whitney's experience in using paneling procedures with internal flows indicates that excessive leakage through the panels prohibits calculation of pressure distributions on the surface with any degree of confidence.

Consequently, it is recommended that PEPSI(G) be refined and enhanced to make it more suitable for the complex geometries of Prop-Fan inlets. A preprocessor must be developed to reduce running times, avoid computational stability problems, and extend the code to more complex geometries. The code must then be calibrated with experimental data and exercised to develop user efficiency and confidence in the results.

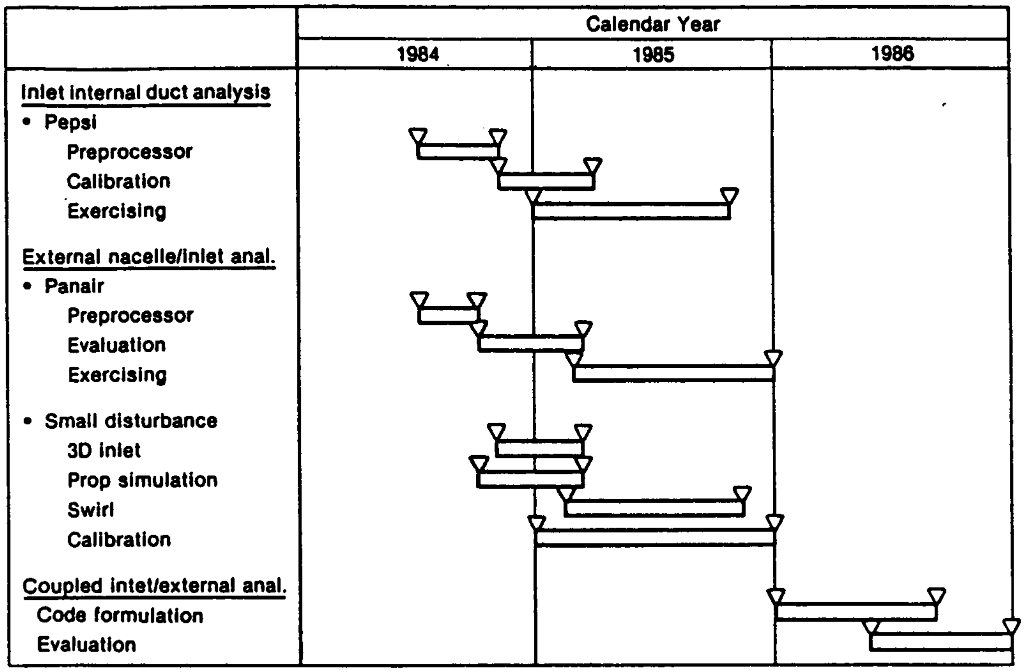

Figure 4.5-20 Analytical Codes for Evaluating Prop-Fan/Nacelle/Inlet Interactions - These codes will be used in conjunction with rig testing to design an efficient Prop-Fan propulsion system. (J27638-201)

Two candidate codes are available for external aerodynamic computations: PANAIR and a Three-Dimensional Small Disturbance Analysis. PANAIR (B868), a pilot adaptation of a code developed by the Boeing Company, is a higher order three-dimensional potential flow (panel) method. It has demonstrated the capability to predict wing/fuselage flow fields at angle of attack. However, PANAIR requires long set-up and run times. A preprocessor can be developed to provide the detailed geometry description required to evaluate the influence of an airplane flow field on the nacelle design, thus reducing processing time. An additional shortcoming of PANAIR is that it does not provide a simulation of the propeller, the associated energy addition, and residual swirl. An effort should be made to establish whether this capability is essential in a Prop-Fan inlet design system.

An alternate system for calculating the external flow field around the inlet is the Transonic Small Disturbance Analysis. This code can model complex geometries quickly and inexpensively. Despite the fact that it is a small disturbance analysis, limited to low turning and thin geometries, it could be well suited to Prop-Fan inlets which are located at large hub radii. With high aspect ratios, the throat height is small and the inlet lip is relatively thin. An added feature of this analysis is the ability to incorporate a simulation of the propeller discharge, including swirl, via an actuator disc analysis. Pratt & Whitney has been using this analysis to simulate the influence of a fan on the inlet and nozzle flow fields of conventional turbofan engines. By

extending the spinner and actuator disc in front of an axisymmetric nacelle, a fair representation of an annular inlet is possible (top of Figure 4.5-21). The lower portion of Figure 4.5-21 shows the resulting calculated pressure distribution over the spinner and around the inlet. The figure also shows the profound impact of the actuator disc on the streamlines as evidenced by the streamtube contraction behind the actuator disc.

Modifications are required before this analysis can be used for chin, bifurcated, and trifurcated inlets. The recommended approach is shown in Figure 4.5-22. The top left of the figure shows the "smile" (chin) inlet that is to be simulated. The ideal analytical model is shown in the center. The top right hand side of the figure shows how, through the use of a permeable nacelle wall over a portion of the circumference, the ideal model can be simulated with Small Disturbance Analysis. The lower portion of the figure shows an actual graphics display of a grid set up to perform these calculations. Additional programming is required to complete the logic. Empirical data must then be used to calibrate the analysis and determine its effectiveness.

Following an assessment of the relative advantages of PANAIR vs Three-Dimensional Small Disturbance Analysis, the appropriate external flow code will be selected and mated to the internal flow (PEPSI) code. By the time this phase has been completed, data will be available from the inlet diffuser rig tests and the Prop-Fan rig tests to calibrate the analyses.

4.5.4.4 Inlet/Low-Pressure Compressor Plan

Previous experience has shown that effective aerodynamic integration of the inlet and engine is a key requirement for a high performance propulsion system. Inlet exit flow conditions must be compatible with major engine design parameters, including static and total pressure profiles, steady state and dynamic distortions, axial Mach number, and surge margin. The consequences of poor inlet and engine compatibility were demonstrated in a recent military turbofan engine program. Inlet/low-pressure compressor integration is even more critical in a Prop-Fan propulsion system.

In the Prop-Fan propulsion system, the inlet aerodynamics behind the multibladed, high tip speed Prop-Fan (with flight Mach numbers in the 0.7-0.8 range) present flow conditions which are much more complex than the conditions encountered in current installations. Transonic exit flow, with the associated swirl component, is expected to produce severe distortion at the inlet front face. The "S" duct inlet contour delivers this distorted flow to the compressor front face with total pressure distortion patterns and approach flow angles significantly different from current experience. Further, the low-pressure compressor, which must include sufficient surge margin to accept the distorted flow and attenuate it to the high-pressure compressor at acceptable levels, will itself incorporate technology features that have yet to be evaluated under these severe conditions. Rig testing is required to establish the actual surge margin loss in the Prop-Fan flow field.

The Inlet/Low-Pressure Compressor Program plan is presented in Figure 4.5-23. There are six major components in the plan: (1) inlet and compressor aerodynamic design, (2) mechanical design, (3) fabrication, (4) assembly and instrumentation, (5) test, and (6) analysis and data reduction. Each part of the plan is discussed in some detail in the following sections.

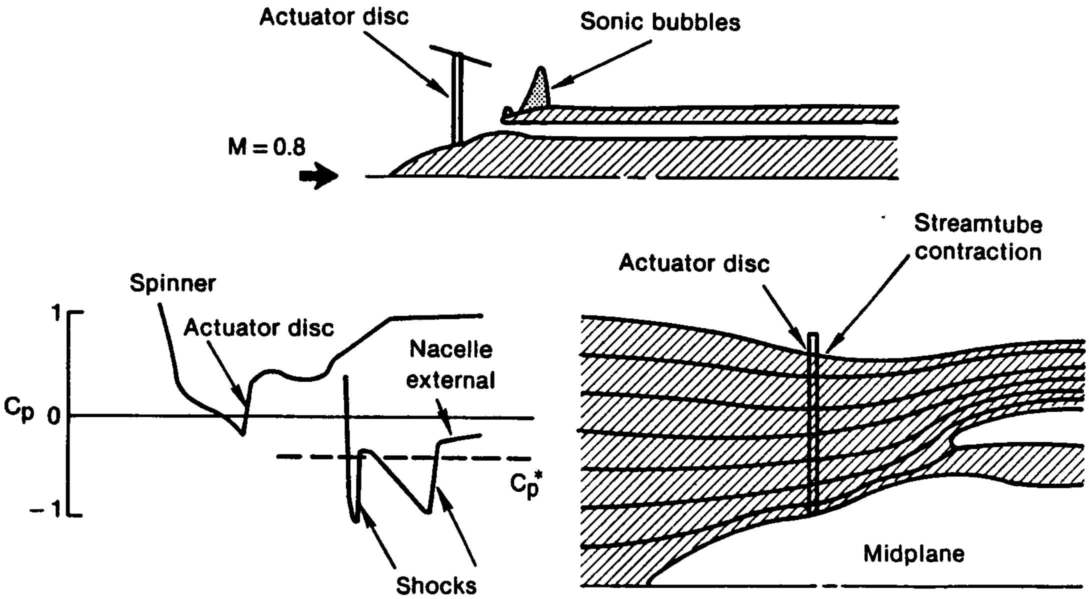

Figure 4.5-21 Turboprop Annular Inlet Flow Field Evaluation - The Transonic Small Disturbance Analysis can be used to generate a fairly accurate representation of an annular inlet. (J27638-103)

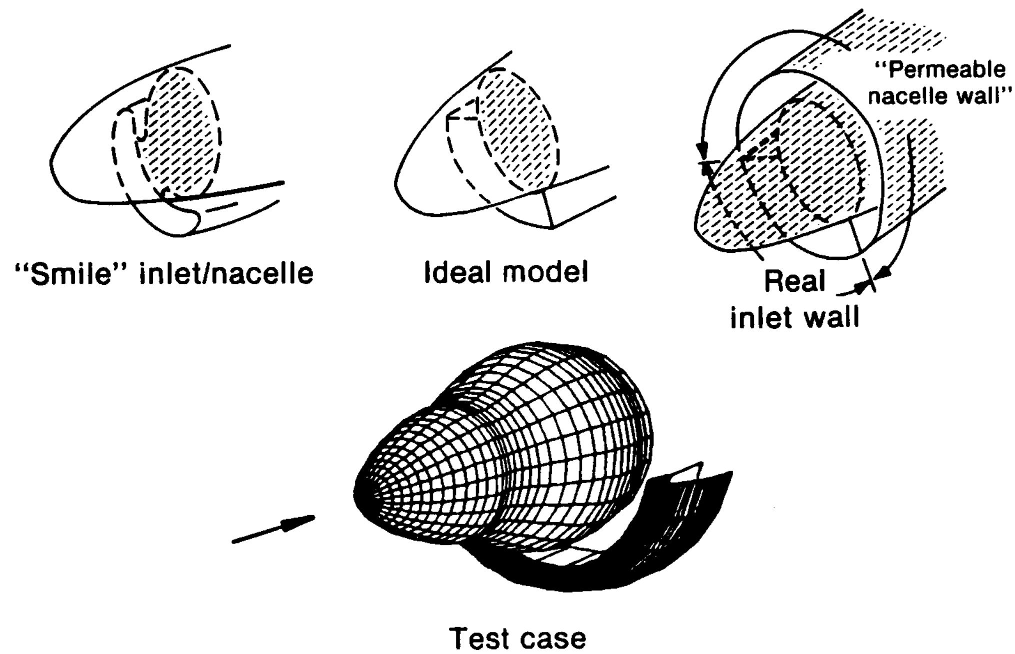

Figure 4.5-22 Turboprop Inlet Model - The Transonic Small Disturbance Analysis can be modified to simulate more complex inlets such as smile, bifurcated and trifurcated configurations. (J27638-97)

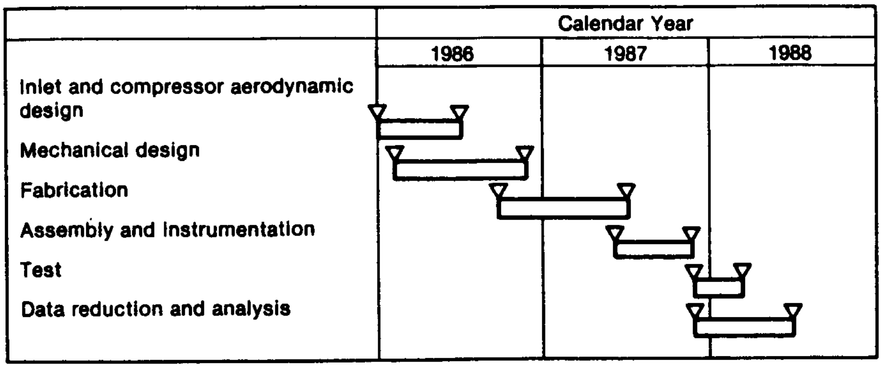

Figure 4.5-23 Inlet/Low-Pressure Compressor Technology Verification Plan Schedule - The program will verify the interactive performance of an advanced turboprop inlet and high-speed low-pressure compressor including the simulated effects of Prop-Fan and nacelle. (J27638-202)

Inlet and Compressor Aerodynamic Design - The rig testing described in Section 4.5.4.2 and the analytical effort discussed in Section 4.5.4.3 will provide the information required to select inlet designs on the basis of propeller/inlet interactions, inlet pressure recovery, and inlet distortion. Design studies will be conducted to select the most promising inlet configuration for the inlet/compressor test. Chin, bifurcated, trifurcated and annular inlets will be considered for both offset and in-line reduction gear systems.

The low-pressure compressor used in the Prop-Fan propulsion system will incorporate advanced technology features that must be adapted to the special inlet boundary conditions of the Prop-Fan system. Inlet radial and circumferential profiles from the Prop-Fan/inlet testing will be incorporated in the design to ensure that low-pressure compressor performance goals are met. Major considerations in low-pressure compressor design are discussed below. Advanced technology features are also described.

Design Considerations - The use of a gear between the Prop-Fan and the low-pressure compressor provides significantly more wheel speed and work capability, but also increases the rotor tip Mach number to a level of 1.3 or greater. These rotor tip Mach numbers are more in the range of front stage high-pressure compressor blades; therefore, a different blade design philosophy is required for the low-pressure compressor in the Prop-Fan propulsion system. High Mach number blade technology, characteristic of current fan blade designs, will be used in the design of the front stages of the low-pressure compressor to produce high work capability without high shock wave total pressure loss. Three-dimensional time marching inviscid solutions of the Euler equations will be combined with interblade boundary layer and shock boundary layer viscous solutions to solve the full flow field aerodynamics to optimize airfoil geometry for maximum efficiency. The three-dimensional flow field solving techniques that will be used in this design have been extensively developed and have been shown to accurately represent the interblade aerodynamics. Agreement of measured and calculated values is excellent in the high Mach number region where minimizing shock loss is essential to good efficiency.

Advanced Technology Features - Controlled Diffusion Airfoils will be used in the design of the remaining low-pressure compressor stages, where Mach numbers are either transonic or high subsonic. In contrast with the high Mach number blade design philosophy, which seeks to control shock waves in order to reduce loss, Controlled Diffusion Airfoils are contoured to reduce loss by eliminating shock waves and avoiding separation of the airfoil surface boundary layers. Controlled diffusion airfoils have demonstrated higher critical Mach number, higher incidence range and higher loading capability than standard airfoils designed to the same aerodynamic requirements.

The low-pressure compressor will also include advanced endwall blading geometry in all rows that will minimize endwall friction, tip clearance loss, stator cavity loss and secondary flow effects. Airfoil sections in the endwall region will be selected to maximize efficiency within the operating range of the low-pressure compressor.

Mechanical Design - The low-pressure compressor test rig will be designed to duplicate the full size aerodynamic flowpath including airfoils, seals, tip clearances, and inner wall flowpath cavities. The static structure will be designed as non-flight (rig) hardware and will incorporate variable vanes in all stages. Existing hardware will be used for bearing compartments and seals wherever possible.

The rotor drum and shafts will also incorporate low cost hardware which meets rig life and structural requirements. Flight hardware aerodynamics will be preserved in all rotating components. The entire rotating structure will be subjected to critical speed analysis to ensure safety throughout the rig operating range. Adapting hardware, such as discharge ducting and drive shaft/coupling, will be designed to adapt to existing test stand interfaces. A complete installation drawing will be generated to show the rig and inlet ducts mounted in the test stand. In addition, a critical speed analysis of the rig and test stand drive train will be conducted to ensure safe operation.

The inlet and compressor will be completely instrumented to provide high response pressures, interstage stator leading edge temperatures and pressures, and inlet duct pressures and temperatures. Circumferential traverse and rotating strain gage instrumentation will also be included.

Fabrication - During this phase of the program, all hardware identified in the mechanical design will be fabricated. Raw material will be procured before the final design is completed, reducing the length and cost of the program. Since the airfoils will be the pacing item in this test, fabrication will be initiated as soon as the designs have been approved. Selected vanes will incorporate machine cuts in which instrumentation will be installed. All other hardware will be fabricated and inspected in compliance with the program schedule. Fixed and traversing performance instrumentation will be calibrated to ensure accurate measurements.

Instrumentation and Assembly - Fixed and traversing instrumentation will be located circumferentially at each axial instrumentation station using an arrangement which ensures that no rake is in the wake of an upstream rake. Early in the assembly phase, sensors will be installed in the airfoils selected for instrumentation. Uniform blade tips and seal clearances will be set by grinding in an assembled condition. All bearing compartments will be leak

checked; oil jet flow will also be tested. The rotor assembly will be dynamically balanced. Variable vane assemblies will be inspected to ensure proper vane stagger and uniformity. In the final assembly, the rotor-stator package will be joined with the inlet case and intermediate case. Quick disconnect blocks will be used for all instrumentation leads.

Test - The assembled test vehicle will be mounted in an existing test stand; a schematic of a typical arrangement is shown in Figure 4.5-24. The test rig will be externally driven by a motor or an engine through a gearbox. Inlet air will be supplied through ducting and will be measured by a suitable nozzle. Air is discharged through ducting and can be controlled by conventional butterfly valves.

A typical rig installation is also shown in Figure 4.5-24. For illustrative purposes, the four-stage compressor is shown in conjunction with a bifurcated inlet. The inlet is protruding into the plenum chamber; flow from the plenum chamber is guided into the inlet using a bellmouth for each duct. Preswirl vanes are used to simulate the exit swirl of the Prop-Fan. Distortion screens and rods (which generate discrete frequencies) are used to simulate the steady state distortions measured from the Prop-Fan/nacelle/inlet tests. The compressor design incorporates several features that reduce costs and facilitate aerodynamic performance optimization in the rig. For instance, the drum rotor design is replaced by a conventional disk design with root mini-cavities. Also, all four stages have variable stators to facilitate stage performance matching.

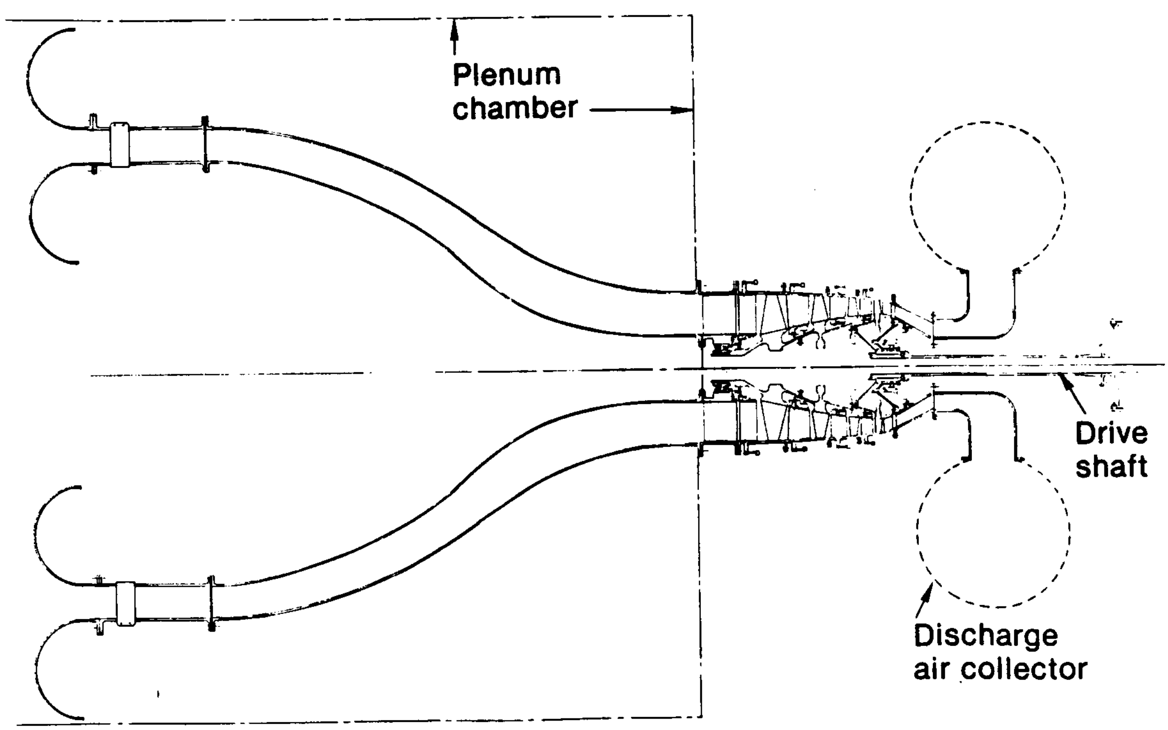

Figure 4.5-24 Low-Pressure Compressor Test Rig - This type of facility will be used to obtain detailed information on low-pressure compressor performance characteristics. (J27638-117)

The data system in the test stand has the capability to measure up to a total of 300 steady state pressures, 158 temperatures and 12 differential pressures. An additional 72 channels can be used for any millivolt signal. The data system is connected to a high-powered Univac computer via telecommunication links. Strain gage instrumentation can be recorded through a 100-channel slip ring.

The test stand also contains a rig Supervisory Control System that allows the engineer conducting the test to control most phases of compressor operation including speed, pressure ratio, vane angle and bleeds. The supervisory control also ensures rig operating safety by monitoring critical performance parameters and taking corrective action to avoid dangerous operating conditions.

The test will be conducted according to an approved test plan. The first phase of the test will consist of a shakedown program to substantiate the mechanical integrity of the test rig throughout the operating envelope and to verify that the instrumentation and data acquisition system are operating properly.

In the second phase, complete performance maps, including vane optimization and stress measurements, will be generated with each of the inlet configurations (e.g., bifurcated and chin). Performance of each of these inlets will also be measured with one or more distortion screens and rods to simulate propeller/inlet interactions over a number of flight conditions.

Data Reduction and Analysis - Two types of computer programs will be used to reduce the raw data for analysis; a conditioning program and a flow field analysis program. A conditioning program will be used to process aerodynamic data and calculate overall and stage performance. This program will also process the data for display. The flow field analysis program will be used to calculate overall, row, and blade element performance, including all vector quantities and parameters necessary for a complete description of compressor performance.

More specifically, the data conditioning deck performs two functions. First, it reduces the electrical signals produced by the instrumentation to engineering units. Second, it applies calibrations and statistical techniques to the resulting data to calculate averaged performance. The results from the first set of calculations are displayed at the test stand, permitting critical parameters to be checked during the test and ensuring that all the instrumentation is operational. The second set of calculations provides a definition of overall performance and stage characteristics, and is used as input to the flow field analysis program.

In the flow field analysis program, inlet and discharge total pressure ratios and temperature ratios are calculated by radially mass-flow-averaging the circumferentially mass-flow-averaged values for the radial locations where data are measured. Static pressures used in determining the average parameters are computed by linearly interpolating values of static pressure measured at the walls. Averaged total pressures and temperatures at each axial location are divided by the compressor inlet values to provide temperature and pressure ratios. The overall performance for any combination of blade rows can be plotted as total pressure ratio, total temperature ratio, and adiabatic efficiency as a function of corrected flow.

Stage performance is presented as a pressure rise coefficient, temperature rise coefficient, and adiabatic efficiency, all as a function of inlet flow coefficient. These coefficients can be used to determine variable stator settings that will optimize the surge margin and efficiency of the compressor.

Traverse data from the compressor exit station wake rakes are presented in the form of wake shapes of pressure, temperature, and efficiency for each radial location traversed. Tabulations of pressure, temperature, and efficiency versus percent stator gap are also provided. Traverse data from the inlet rakes are presented in the form of radial total pressure variations of each circumferential traverse position from which overall annular distortion maps will be produced.

Input for the flow field analysis program includes values of inlet corrected flow, corrected speed, inlet total pressure and total temperature, and radial distributions of total pressure ratio, total temperature ratio, and reference static pressures from selected instrumentation planes.

The detailed overall, stage, blade and vane row, and blade element data are analyzed relative to the design goals and major deviations are noted. Blade changes designed to improve the overall performance of the compressor are identified and evaluated using computer programs. These changes could be made to improve the performance match of front end stages to the performance of back end stages, to correct spanwise pressure profile defects, or to implement local changes to correct blade element traverse incidence conditions. The differences between design objectives and test results will be analyzed to determine the effect and magnitude of the Prop-Fan inlet interactions and to formulate a set of Prop-Fan low-pressure compressor design ground rules that recognize the special requirements of these interactions. Variations in the performance of the two types of inlets will also be analyzed to formulate special design requirements and to identify characteristics of the inlet system which have either favorable or deleterious effects on the compressor. A preferred inlet system will be recommended on the basis of overall system and compressor performance data.

4.5.5 Small-Size High-Pressure Compressor

Engines of the 1990's and beyond are expected to have higher overall pressure ratios than engines currently in service. To achieve these higher pressure ratios, blade heights in the rear stages of the high-pressure compressor will be reduced. These small blades are susceptible to erosion and early loss of performance, a condition that is aggravated by the smaller size core of advanced turboprop engines relative to turbofan engines of equal thrust. For example, the blade lengths for the 12,000 shp advanced turboprop engines described in Section 4.3 are approximately 1.5 cm (0.6 in).

Engine configuration studies conducted in Task II (described in Section 4.2) highlighted the good deterioration characteristics of centrifugal/mixed-flow compressor stages after approximately 3500 cycles. (As indicated previously, a mixed-flow compressor stage is similar to a centrifugal compressor stage, except that the flow at the exit has an axial velocity component in addition to a large radial component.) In addition, the studies demonstrated the low maintenance cost of centrifugal/mixed-flow stages. The improved fuel burn characteristics after 3500 cycles, coupled with reduced maintenance costs, contribute to significantly lower direct operating costs for a high-pressure compressor configuration with centrifugal/mixed-flow stages.

Another advantage of centrifugal compressors in small engines is the reduced rotor length relative to axial compressors. This is especially helpful in solving rotor critical speed problems that are more crucial in smaller engines.

Studies conducted under the Energy Efficient Engine program have also confirmed that a centrifugal or mixed-flow compressor would be an attractive alternative to an all-axial compression high-pressure compressor for an advanced turbofan engine.

A high-pressure compressor for an advanced turboprop or turbofan engine could, therefore, include a number of axial compressor stages followed by a single centrifugal or mixed-flow compressor stage of moderately high pressure ratio. The number of axial compressor stages is dictated by the work potential of the centrifugal or mixed-flow stage. Work potential is determined by the tip speed limit, which is set by structural considerations at the compressor exit. The extremely high temperatures encountered at the compressor exit increase the difficulty of creating an effective design.

A systematic analytical/test program is required to determine the relative merits of using all-axial rear stages vs centrifugal or mixed-flow rear stages in advanced engines.

4.5.5.1 Objectives and Benefits

The objectives and benefits of the small-size, high-pressure compressor technology program are summarized in Table 4.5-IV.

TABLE 4.5-IV
OBJECTIVES AND BENEFITS
High-Pressure Compressor Technology Program

Objectives:

- o Verify the performance of a centrifugal or mixed-flow compressor in order to permit use in the rear stages of an advanced turboshaft engine. This configuration will improve deterioration characteristics, reduce maintenance cost, and reduce acquisition cost relative to axial compressor stages.

- o Develop new advanced analytical design codes.

Benefits:

- o Compressor efficiency loss decreased by 1.5 percentage points after 3500 cycles of operation.

- o Engine maintenance cost reduced approximately 12%.

- o Engine acquisition cost reduced approximately 8%.

- o Compressor/diffuser length reduced approximately 25%.

- o All of these benefits achieved with a maximum 5% increase in engine weight.

4.5.5.2 Program Plan

Three candidate configurations for the rear stages of the high-pressure compressor were considered in the APET Definition Study: axial stages, a mixed-flow stage, and a centrifugal stage. These concepts should be compared on the basis of efficiency, weight and cost and the optimum configuration should be selected for the Prop-Fan propulsion system.

The flow diagram for the small-size high-pressure compressor program is shown in Figure 4.5-25. The schedule for the high-pressure compressor technology program is shown in Figure 4.5-26. The individual phases of the program are discussed below (the section numbers for each phase are specified in the figure).

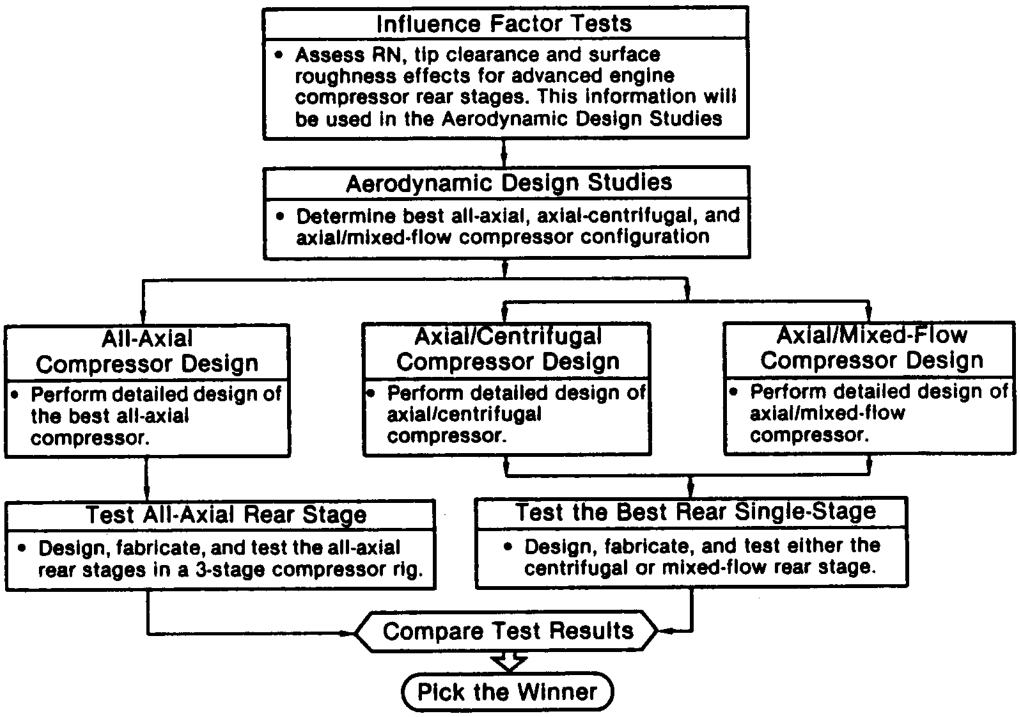

Figure 4.5-25 High-Pressure Compressor Program Flow Diagram - The program is designed to identify the rear stage compressor configuration that will provide optimum efficiency and performance retention. (J27638-204)

4.5.5.3 Influence Factor Tests

A series of tests will be conducted with an existing three-stage rig to assess the effects of variations in Reynolds number, tip clearance and surface roughness from current engine levels to the range of the Prop-Fan engine. The parameters are compared in Table 4.5-V.

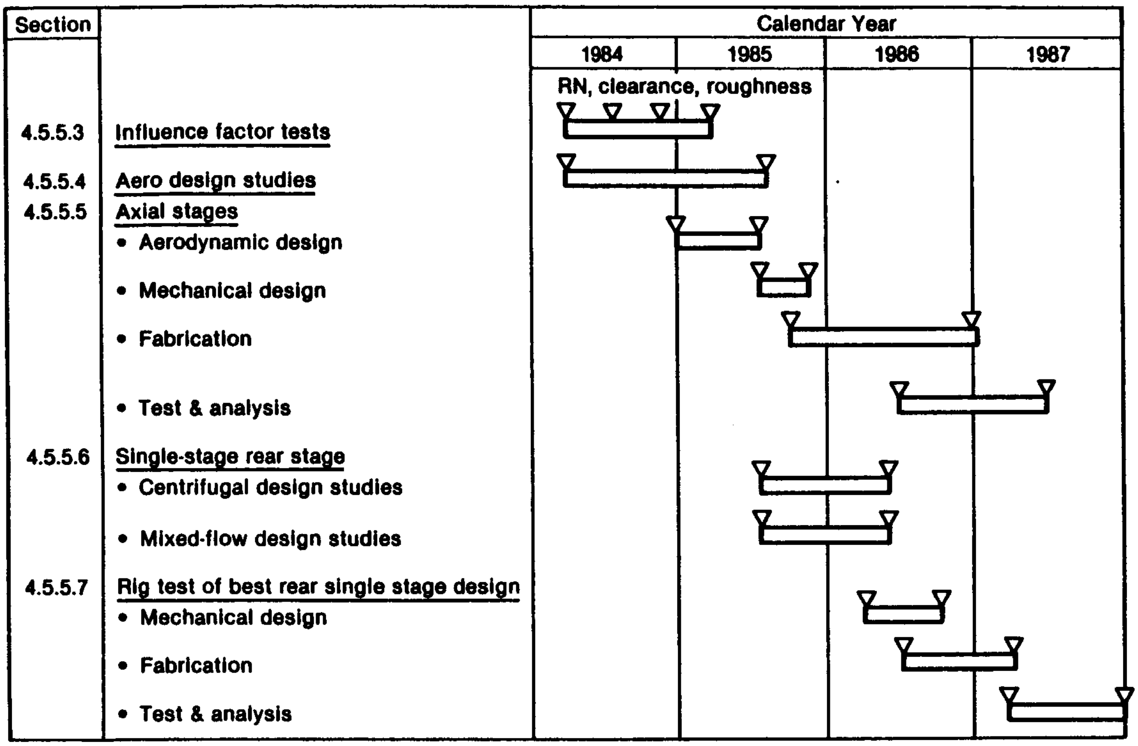

Figure 4.5-26 High-Pressure Compressor Technology Verification Plan - This five-phase program will identify the best rear stage configuration for the high-pressure compressor of the Prop-Fan propulsion system. (J27638-191)

TABLE 4.5-V
COMPARISON OF CURRENT ENGINE HIGH-PRESSURE COMPRESSOR PARAMETERS TO PROP-FAN HIGH-PRESSURE COMPRESSOR PARAMETERS

	State of the Art Rear Stages	Estimated APET Rear Stages
Mach Number (Rotor/Stator Average)	0.63	0.6
Reynolds Number x 10^{-5}	10.3	5.0
Tip Clearance/Span	1.0%	2.0%
Blade Span, cm (in)	3.3 (1.3)	1.65 (0.65)

The Pratt & Whitney closed loop test facility at the United Technologies Research Center is well suited to conduct these tests; it provides the pressurization capability to vary Reynolds number by a factor of over 3.0. This facility, which is shown in Figure 4.5-27, can be used to test full scale core compressor middle and rear three-stage rigs such as the rig shown in Figure 4.5-28. Flowpath, blading and stator cavities are representative of current practice for commercial engines. The type and number of blades in this core compressor three-stage rig can be modified to meet the requirements of the APET High-Pressure Compressor Program. The three critical parameters essential to compressor performance and performance retention - tip clearance, Reynolds number, and airfoil surface roughness - will be evaluated over the range of values shown in Table 4.5-VI.

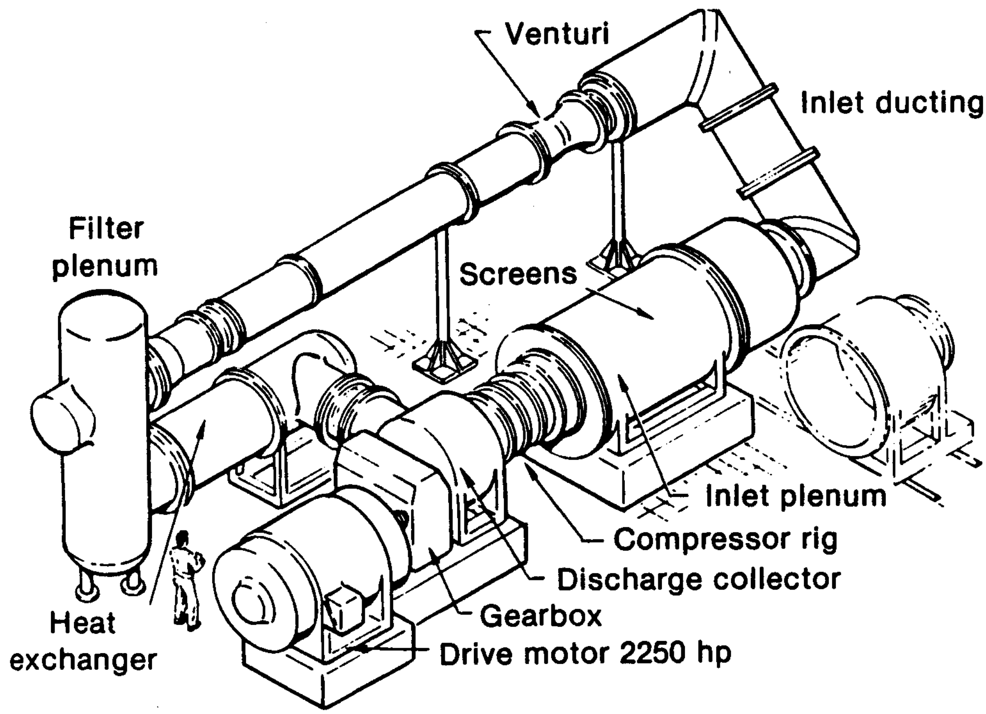

Figure 4.5-27 United Technologies Research Center Closed Loop Compressor Test Facility - This type of facility is well suited to conducting compressor rig tests. (J27638-110)

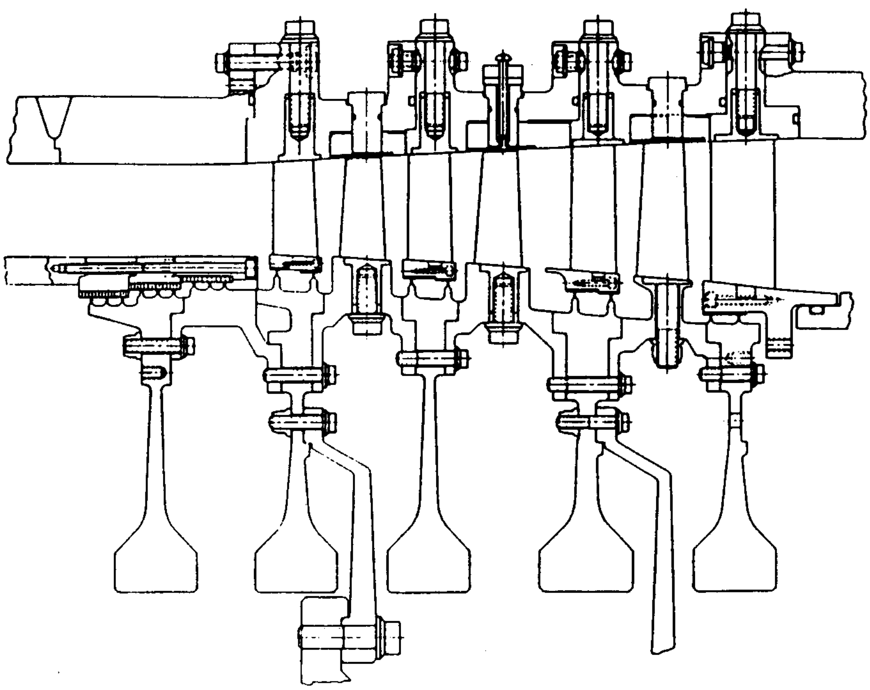

Figure 4.5-28 Typical Three-Stage Compressor Rig - This compressor rig will be used to test the effects of variations in Reynolds number, tip clearance, and surface roughness on high-pressure compressor performance. (J27638-109)

TABLE 4.5-VI
SUGGESTED THREE-STAGE RIG FOR PROP-FAN HIGH-PRESSURE COMPRESSOR INFLUENCE FACTOR PROGRAM

Test Number	Configuration	Tip Clearance	Reynolds Number	Airfoil Surface Roughness
		Span	$\times 10^{-5}$	Arithmetic Avg $\times 10^{-6}$
Previously Tested	Baseline	1.0%	3.8-11.6	10-20AA
1	Increased Rotor Clearance	2.0	3.8-11.6	10-20AA
2	Increased Surface Roughness	2.0	3.8-11.6	4000 plus hours in-service simulation

Instrumentation in the rig will include: (1) inlet and discharge pole rakes to establish radial total pressure and total temperature profiles, (2) stator leading edge total pressure and total temperature sensors to determine stage radial profiles and matching, and (3) stator surface static pressure taps to determine airfoil surface Mach number and to evaluate the accuracy of blade design techniques in this regime. The detailed information provided by the static pressure measurements on the airfoil surface will be coupled with overall performance results and compared to design objectives. Design techniques will be modified as required to accurately represent conditions in the Prop-Fan engine environment. These improved design tools will then be used in the aerodynamic design studies and in the final design.

4.5.5.4 Aerodynamic Design Studies

Aerodynamic design studies will be conducted to identify the best all-axial, axial/centrifugal and axial/mixed-flow configurations for advanced engines. Compressor efficiency, weight, cost and performance retention will be considered in the selection of a final configuration in each category. The impact of advanced technology on the efficiency and surge margin potential of each configuration will be considered in the evaluation. Results of the high-pressure compressor tests described in Section 4.5.5.3 will provide a realistic baseline for performance evaluation and influence factors used to assess the effects of Reynolds number, tip clearance and surface roughness. An existing meanline performance prediction system, updated to include the test results, will be used to select the best configurations. Flowpaths, airfoil counts and preliminary weight and cost estimates will be prepared for each configuration.

4.5.5.5 Axial Stages

This phase of the program includes aerodynamic design of the all-axial compressor, followed by mechanical design, fabrication and testing of the rear stage compressor rig.

Aerodynamic Design - A detailed full span aerodynamic design will be conducted for the final axial configuration; a complete specification of blade geometry will be developed from the design. Blade design will be optimized to produce maximum efficiency within the surge margin constraints at design and off-design conditions. Blade technology will include full three-dimensional high Mach number concepts developed for current turbofans in the front stages and advanced controlled diffusion/improved endwall geometry in the remaining stages. Airfoil boundary layer transition and separation criteria will be modified as required, based on the results of the influence factor tests. These modifications will produce airfoil geometry that accurately reflects the operating conditions in the Prop-Fan propulsion system. Endwall geometry will be contoured to maximize performance at the required level of rotor tip clearance and stator seal clearance.

In the final phase of the design, the potential for replacing several of the rear stages of the axial compressor with a centrifugal or mixed-flow compressor will be evaluated. The rear axial compression stages will be tested in the high speed closed loop facility and will serve as a baseline for evaluating the performance potential of the single mixed-flow or centrifugal compressor stage.

Mechanical Design of Rear Stage Compressor Rig - Three stages of airfoils will be designed for the rear stage compressor rig. The static structure will be designed as non-flight (rig) hardware. Existing hardware will be used for the bearing compartments and seals wherever possible.

The rotor and shafts will also incorporate low cost hardware that meets rig life and structural requirements. The entire rotating structure will be subjected to critical speed analysis to ensure safety throughout the rig operating range. An installation drawing will be generated showing all the rig/test stand interfaces.

Provisions will be made for complete instrumentation in the test rig including major station inlet and discharge probes, interstage leading edge instrumentation, and high response instrumentation.

Fabrication of Rear Stage Compressor Rig - During this phase of the program all hardware identified in the mechanical design will be fabricated. Raw material will be procured before the final design is completed, reducing program cost and length. Since the airfoils will be the pacing item in this test, machining will be initiated as soon as the design is approved. Selected vanes will incorporate machine cuts in which instrumentation will be installed. All other hardware will be fabricated and inspected in compliance with the program schedule. All fixed and traversing performance instrumentation will be calibrated to ensure accurate measurements.

Test of Rear Stage Compressor Rig - A series of three tests will be conducted with a newly fabricated three-stage rig to assess the effects of Reynold's number, tip clearance and airfoil surface roughness. These tests will be conducted in the closed loop test facility shown in Figure 4.5-27.

The program will begin with a shakedown to establish the mechanical integrity of the test rig and to verify the operation of instrumentation and data reduction equipment. The performance test will consist of three separate rig configurations designed to evaluate:

Tight tip clearance blading with rough surface finish airfoils
Loose tip clearance blading with rough surface finish airfoils
Loose tip clearance blading with smooth surface finish

After the second and third rig configuration tests the rigs must be removed from the test stand and the hardware modified. In each of the three configuration tests, a series of speedlines will be run from wide open throttle to surge at a number of different Reynold's number settings.

4.5.5.6 Centrifugal and Mixed-Flow Rear Stage Design Studies

Preliminary studies have shown the potential economic advantage and improvement in deteriorated performance that can be obtained by replacing the rear stages of high overall pressure ratio engines with a centrifugal or mixed-flow compressor. This advantage stems primarily from replacing several axial compressor blades and vanes with a single, rugged and less costly impeller and diffuser with improved resistance to performance deterioration.

Previous studies have shown small differences in the performance of axial and centrifugal compressor configurations. However, performance depends to a great extent on the level of tip clearance that can be maintained in the axial compressor and the rotational speed of both configurations. Advanced technology concepts for axial compressors, such as improved endwall geometry, have the potential to improve the performance of axial systems, even under the adverse conditions encountered in the rear stages of advanced turboprop engines. The recommended technology programs for centrifugal compressors would also be directed toward improving performance at the low specific speeds of Prop-Fan system applications. The impact of advanced design and manufacturing technologies will be considered in the centrifugal and mixed-flow stage aerodynamic and mechanical design studies in order to provide a realistic assessment of the direct operating costs of these two configurations, as well as an objective comparison with the axial compression stages which these configurations would replace.

This design work will form the basis for selecting the most promising single-stage configuration, centrifugal or mixed-flow, for comparison with an optimized axial compression system.

Centrifugal Rear Stage Design Studies - Full three-dimensional flowfield definition computer programs and advanced high Mach number airfoil sections will be used to design the centrifugal compressor for peak performance at operating conditions characteristic of the Prop-Fan propulsion system. The impact of using pipe and vaned diffusers on engine direct operating cost will be evaluated.

Impeller endwall geometry will be optimized for the low specific speed and high wall friction encountered in the Prop-Fan propulsion system. Full blading geometry specifications and flowfield definition will be produced.

The impact of advanced manufacturing technology on engine weight, cost, and structure will be considered in the mechanical design of the centrifugal impeller and diffuser. The flight hardware will be defined in sufficient detail to permit realistic evaluation of weight, cost and performance relative to the axial and mixed-flow rear stages.

Mixed-Flow Rear Stage Design Studies - The mixed-flow configuration is being considered in these studies because it represents a reasonable compromise between the large diameter required for the centrifugal compressor and the large number of airfoils required for the axial compressor. Aerodynamic design techniques developed for axial compressors, including high Mach number airfoil section theory and three-dimensional definition, will be applied to the mixed-flow configuration. Structural analyses will be conducted to assure that stress and life cycle requirements for the Prop-Fan propulsion system are satisfied.

4.5.5.7 Rig Test of the Best Rear Single-Stage Design

The most effective single-stage rear compressor configuration will be selected based on the design studies. A rig will then be designed, fabricated and tested as described below.

Rig Mechanical Design - A compressor rig with integrated axial and impeller stages will be designed for the best rear stage configuration. The static structure will be designed as non-flight (rig) hardware. Existing hardware will be used for the bearing compartments and seals wherever possible.

The rotor and shaft will also incorporate low cost hardware designed to satisfy rig life and structural requirements. Special attention will be given to the structural analysis of impeller type rotors. The entire rotating structure will be subjected to critical speed analysis to insure safety throughout the rig operating range.

Provisions will be made for complete instrumentation of the test rig including major station, inlet and discharge probes, interstage leading edge instrumentation and high response instrumentation.

Fabrication - The latest cost-effective manufacturing techniques will be used in fabricating the centrifugal or mixed-flow compressor impeller.

The impellers will be manufactured by isothermal forging to near net shape. One of the key elements is the use of the GATORIZING® process, which Pratt & Whitney's Government Products Division has used to forge a rotor to near net shape. In addition, a differential heat treatment, producing a fine grain structure in the rotor hub and coarse equiaxed grains in the blade region, has been developed in a subscale component.

Test - A performance test will be conducted using a test rig incorporating the best rear stage configuration. The compressor rig will be tested in the closed loop facility shown in Figure 4.5-27.

The test program will consist of a shakedown program to substantiate the mechanical integrity of the rig throughout the operating envelope and to verify proper operation of the instrumentation and data acquisition system.

In the performance testing, a series of speed lines will be run from wide open throttle to surge. Readings form steady state performance instrumentation will be recorded throughout the operating range of the compressor.

4.5.6 Recommended Engine/Aircraft Integration Studies

A number of technical considerations has identified during Task II (Cycle Optimization and Engine Configuration Selection) and Task III (Propulsion System Integration) that affect both the Prop-Fan propulsion system and the aircraft on which it will be mounted. After reviewing the Prop-Fan Propulsion System Integration Package, the airframe manufacturers participating in the APET Definition Study and Pratt & Whitney identified several key issues which should be addressed in joint engine/aircraft integration studies. These issues encompass potential "barrier" technologies, which, if not resolved in a timely manner, could prevent a Prop-Fan powered aircraft from being introduced to service by the 1992 target date.

These recommendations have been consolidated in four major engine/aircraft integration programs, listed in order of importance:

1) Free vs Non-Free Power Turbine Engine/Aircraft Study
2) Engine/Aircraft Propulsion System Mounting Integration
3) Engine/Aircraft Heat Rejection Study
4) Integrated Propulsion System/Aircraft Control

A brief description of each program follows. Individual tasks are defined and preliminary program schedules are presented. Preliminary plans for these programs have been discussed with the airframe manufacturers. Once NASA specifies which engine/aircraft integration studies should be conducted, program content will be defined in detail and final program plans will be prepared. Estimates of the costs of these studies are provided in a separate proprietary document.

4.5.6.1 "Free" vs "Non-Free" Power Turbine Engine/Aircraft Integration Trade Study

The overall objective of this study is to determine whether there are fundamental differences in the performance of the two-spool, all-axial compression, "non-free" power turbine engine (STS678) and the three-spool, axial/centrifugal compression, "free" power turbine engine (STS679) at critical aircraft operating conditions. In Task II (Engine Configuration Selection), both engines were found to be viable candidates for a Prop-Fan propulsion system. Steady state and transient operating characteristics will be evaluated in the free vs nonfree power turbine engine/aircraft integration study.

To illustrate the type of conditions that will be evaluated, Figure 4.5-29 shows Prop-Fan speed as a function of power setting for the "free" and "non-free" power turbine engines, as well as the typical steady state approach power requirement. With the non-free power turbine engine, approach power is about 70%-80% of the takeoff speed of the Prop-Fan; with the free power turbine engine, approach power is 100% of the takeoff speed of the Prop-Fan. In the event of a "go-around" or aborted approach, Prop-Fan speed must be accelerated up to the takeoff thrust level with the non-free power turbine engine. With the free power turbine, the Prop-Fan is already operating at full speed, and the required power can be achieved by simply changing the pitch of the Prop-Fan blades. This example demonstrates the contrasting transient thrust response characteristics of the two engines.

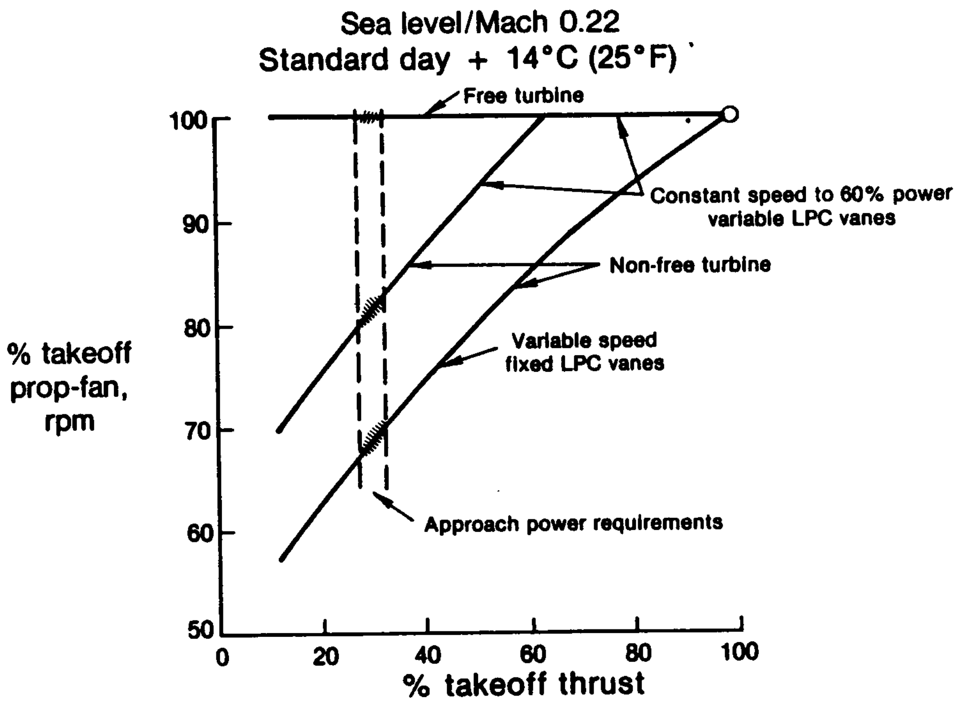

Figure 4.5-29 Operating Characteristics of the "Free" and "Non-Free" Power Turbine Engines at Approach - With the non-free power turbine engine, approach power is about 75% of Prop-Fan takeoff speed; with the free power turbine engine, approach is made at full speed. (J27638-217)

However, this example depicts only one of the many operating conditions that must be considered in evaluating the dynamic response characteristics of the two engines. In the free vs non-free power turbine engine/aircraft integration study, the differences in the responses of the two engines will be quantified, and data will be provided to the airframe manufacturers which will permit them to evaluate the impact of these differences on the Prop-Fan powered airplane.

In addition to evaluating the differences in transient operation between the two engines, two methods of improving the flexibility of the non-free turbine engine will be evaluated (see Figure 4.5-29). In the first method, variable vanes are incorporated in the low-pressure compressor in order to maintain a constant Prop-Fan speed over the 60%-100% thrust range. In the second method, the geometry in the low-pressure compressor is fixed, while Prop-Fan speed is varied over the engine operating range. The impact of operating the Prop-Fan at constant and variable speeds will be evaluated on the basis of performance, noise, and transient response.

The schedule for the free turbine vs non-free turbine engine/aircraft integration study is shown in Figure 4.5-30. A description of the individual tasks follows.

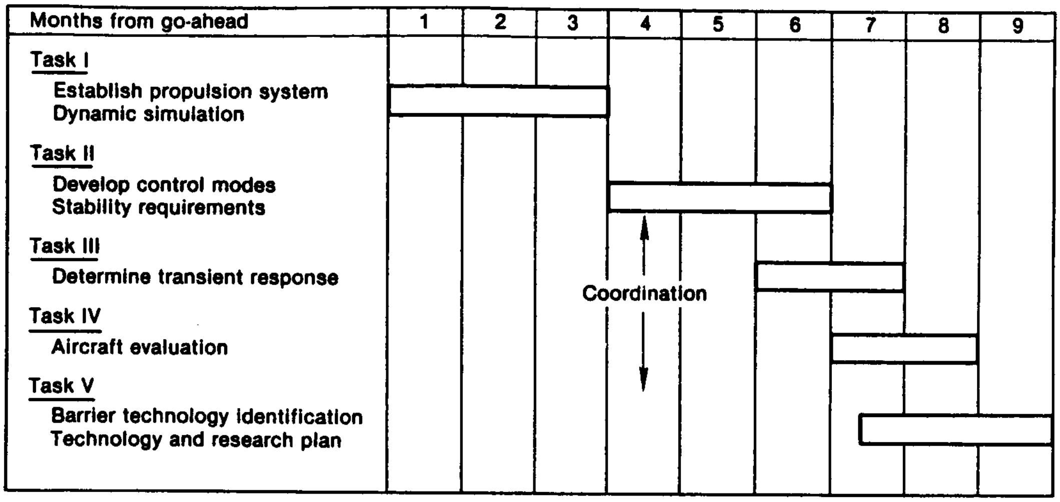

Figure 4.5-30 Free vs Non-Free Power Turbine Engine/Aircraft Integration Study Schedule - The five technical tasks in this program will aid in evaluating the operating characteristics of the free and non-free power turbine engines. (J27638-216)

Task I - Propulsion System Dynamic Simulation

The dynamic operating conditions of the free power turbine and non-free power turbine engines will be simulated. The models will provide a quantitative representation of the characteristics of the propulsion systems during transient operation. Dynamic simulations will be developed by modeling engine rotor, Prop-Fan and gearbox inertia characteristics, Prop-Fan blade angle and aerodynamic characteristics, as well as the effects of using variable vanes in the low-pressure compressor of the non-free power turbine engine.

Task II - Control Modes and Stability Requirements

Methods of operation for the propulsion system electronic control will be defined under a variety of operating conditions. These specifications will ensure that appropriate thrust response is obtained from the free turbine and non-free turbine engines through effective management of fuel flow, compressor geometry, Prop-Fan blade pitch, and active clearance control (if required). The dynamic propulsion system simulations developed in Task I will aid in defining the control modes. Compression system stability requirements will also be a major factor in the control mode definition, ensuring safe operation under a variety of conditions.

The impact of key assumptions about engine components, such as the weight and inertia of centrifugal compressors, on the transient response characteristics of the two engine configurations will also be assessed.

Task III - Transient Response

The control modes defined in Task II will be used to determine the transient response characteristics of the free and non-free power turbine engines at critical operating conditions. The conditions to be evaluated will be selected by Pratt & Whitney and the participating aircraft manufacturer and approved by the NASA Program Manager. The "missed approach" situation described earlier is typical of the conditions that will be studied.

Steady state engine operations will be evaluated with fixed and variable speed Prop-Fans in order to assess the impact of different methods of operation on performance and noise.

Starting and windmilling engine operating conditions, which were covered during the APET Definition Study, will be excluded from this effort.

Task IV - Aircraft Evaluation

The impact of engine steady state and transient response characteristics on airplane operation will be assessed. This study will be conducted by the airframe manufacturer.

Task V - Research and Technology Plan

A detailed Research and Technology plan will be prepared to verify key technologies identified in this study and to ensure timely response to issues that could delay certification of a Prop-Fan powered aircraft.

4.5.6.2 Engine/Aircraft Propulsion System Mounting Integration

This study will address two key propulsion system/aircraft integration issues: (1) use of vibration isolation devices to damp propulsion system induced vibrations, avoiding transmission through the wing to the fuselage, and (2) establishment of key geometry constraints to prevent wing flutter resulting from propulsion system loads. All of the airframe manufacturers participating in the APET Program indicated that this study should be accorded high priority.

The schedule for the propulsion system mounting study is shown in Figure 4.5-31. A description of the individual tasks follows.

Task I - Analytical Model

An analytical model of the propulsion system and wing structure will be generated using the integrated engine mount system developed in Task III of the APET Definition Study. The model will be used to identify the optimum location of the propulsion system for a given wing structure. The model will also provide spring rate and damping requirements for vibration isolation devices.

Task II - Evaluation of Alternate Mounting Concepts

Conceptual designs will be prepared for up to three different propulsion system mounting arrangements.

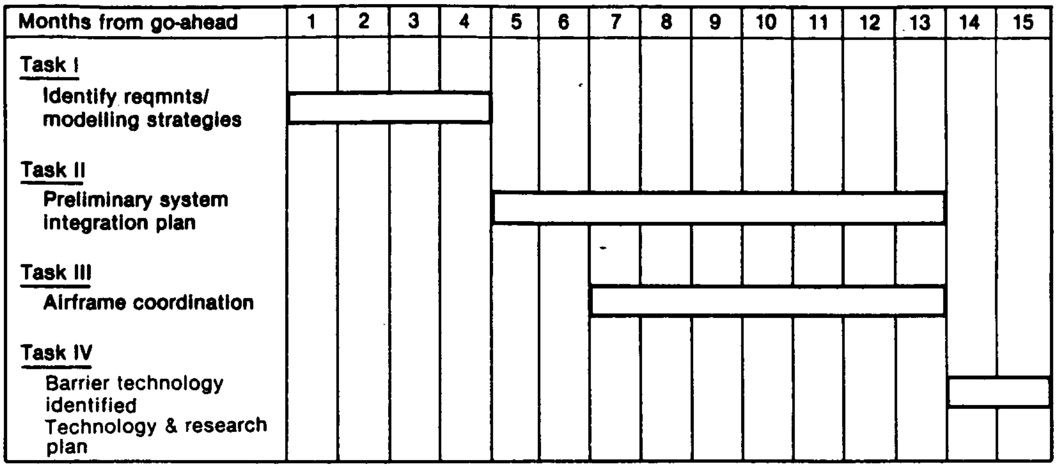

Figure 4.5-31 Propulsion System Mounting Study - This study will address key propulsion system mounting issues: vibration isolation and wing flutter. (J27638-218)

Task III - Cost/Benefit Analysis

An economic analysis will be conducted to select the best mounting system for a Prop-Fan powered aircraft.

Task IV - Preliminary Design

A preliminary design will be prepared for the mounting system selected in Task III. The preliminary design will specify vibration isolation requirements, define propulsion system dynamic characteristics, provide a method for structurally integrating the reduction gearbox with the engine, and identify modular maintenance concepts for the mounting system.

Task V - Research and Technology Plan

A detailed Research and Technology plan will be prepared to verify key technologies identified in this study and to ensure timely response to issues which could delay certification of a Prop-Fan powered aircraft.

4.5.6.3 Engine/Aircraft Heat Rejection Study

A critical characteristic of the Prop-Fan propulsion system is the significant amount of heat generated in the oil systems relative to current turbofan engines producing comparable thrust. In fact, the heat in the oil systems is so great that the systems cannot be effectively cooled by the fuel consumed in the engine. To alleviate this problem, several air/oil cooling concepts were evaluated in the APET Definition Study. However, these systems were eliminated from consideration due to significant weight, cost and/or performance penalties.

A system in which the fuel in the aircraft tanks is used as a heat sink and a supplementary air/oil cooler is provided for auxiliary operation was finally selected. Since this is a relatively new concept, the airframe manufacturers recommended that additional studies be conducted to assess the relative merits of the system. The engine/aircraft heat rejection program shown in Figure 4.5-32 will address the key technical considerations for this system. A description of the individual tasks in the program follows. At the outset of the study, the total heat rejection for the propulsion system (engine, gearbox, and Prop-Fan) and aircraft (environmental systems, etc.) will be determined.

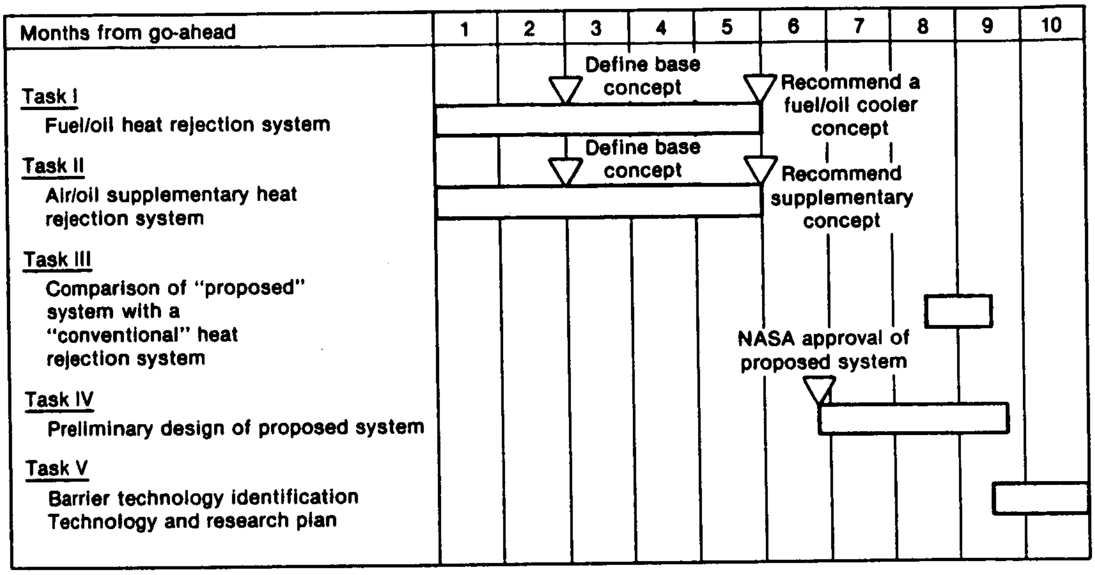

Figure 4.5-32 Engine/Aircraft Heat Rejection Study - This program will address key technical considerations for a keat rejection system for a Prop-Fan powered aircraft. (J27638-219)

Task I - Fuel/Oil Heat Rejection System

The use of fuel in the aircraft tanks as a heat sink will be evaluated. Major considerations include:

- Tank capacities, flight utilization and cooling rates
- Engine-to-tank line heat loss definition
- Tank location and fuel management benefits
- Fuel deicing
- Heat rejection loads for aircraft subsystems

Task II - Supplementary Air/Oil Cooler System

The supplementary air/oil cooler system used for auxiliary operation will be evaluated. Major factors include the design of low drag inlets and exhaust, and definition of the sizing, location, and operational requirements of the system.

Task III - Comparison with Conventional Heat Rejection System

The fuel/oil heat rejection system with supplementary air/oil cooler will be compared to a conventional heat rejection system on the basis of cost and performance characteristics.

Task IV - Preliminary Design

With the approval of the NASA Program Manager, a preliminary design will be developed for the basic fuel/oil heat rejection system and the supplementary air/oil cooler.

Task V - Research and Technology Plan

A detailed Research and Technology plan will be prepared to verify key technologies identified in this study and to ensure timely response to issues which could delay certification of a Prop-Fan powered aircraft.

4.5.6.4 Integrated Propulsion/Aircraft Control Study

The control system identified in the APET Definition Study (Section 4.3.1.6) is an advanced, dual channel, full authority digital electronic control incorporating electronic circuitry, fiber optics, and dual redundancy in the vital control paths. The control provides independent control of Prop-Fan blade pitch (synchrophasing, etc.), engine speed/power setting, automatic control in steady state and transient operation for forward/reverse thrust, and protective measures for limiting torque, temperature, overspeed, and possible system fault (Prop-Fan feathering, windmilling, etc.). However, this control system is conceptual in nature. A program is required to identify and verify the technology for an effective, integrated control system for a Prop-Fan powered aircraft by 1988.

Experience has shown that a comprehensive plan for integrating the components in the propulsion system electronic control with other aircraft systems can lead to major improvements in safety, performance and cost. The primary emphasis in this study will be a thorough examination and understanding of the requirements posed by an integrated control system concept. A fundamental approach to control system evaluation is the use of models to qualitatively assess alternate configurations and design approaches. This study will identify the methodology for designing a high quality integrated control system and present a plan for addressing barrier technology issues.

The schedule for the integrated propulsion system/aircraft control study is shown in Figure 4.5-33. A description of the individual tasks follows.

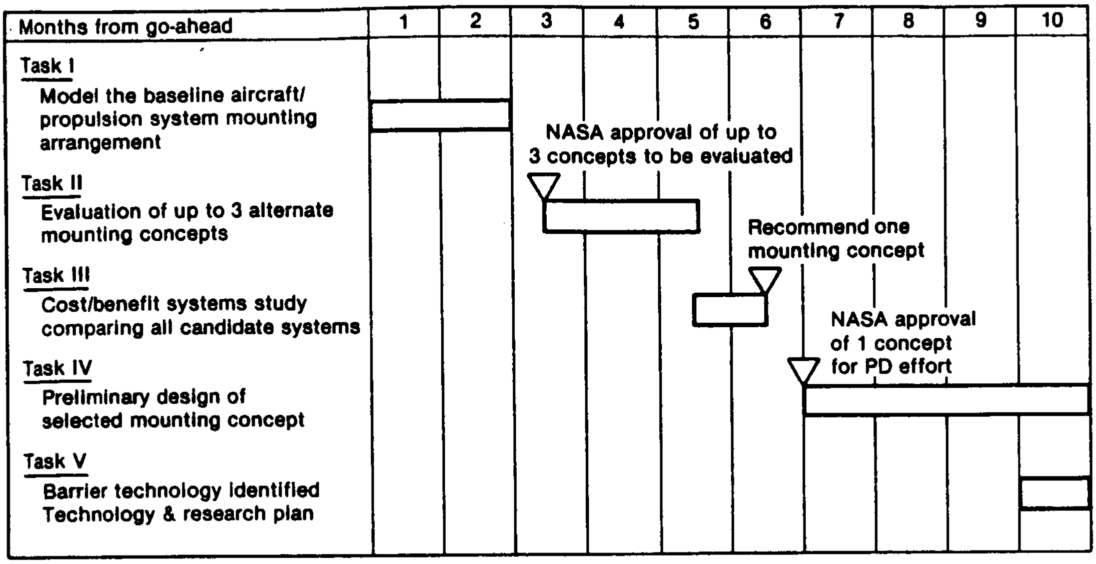

Figure 4.5-33 Integrated Propulsion System/Aircraft Control Study Schedule - In this study, technology for an integrated control system for a Prop-Fan powered aircraft will be developed and verified. (J27638-220)

Task I - Identify Requirements and Modeling Strategies for Control System Design and Evaluation

A Systems Requirement Document will be generated to define specific functional and performance requirements for the propulsion and aircraft systems. This document will provide a focal point for identifying features to be incorporated in the electronic control. Advanced modeling techniques will be used to evaluate various control system concepts and design approaches on the basis of cost, weight, and maintenance considerations.

Task II - Preliminary System Integration Plan

The Systems Requirement Document (SRD) will be used to create a preliminary System Integration Plan. This plan will ensure that the technology features identified in Task I are incorporated in the integrated electronic control for the Prop-Fan propulsion system. Experience with control systems for the PW2000 and PW4000 engine families will provide significant input to the integration plan.

The System Integration Plan covers seven major factors in control system design:

- o Create control modes and laws that satisfy system performance requirements.
- o Define component reliability characteristics that ensure safety and maintainability.
- o Describe the physical arrangement of the components and the interface requirements.
- o Coordinate aircraft interface requirements with airframe manufacturers and translate them into control system requirements.
- o Define a fault accommodation system that meets reliability requirements.
- o Provide maintenance capability that minimizes end-user costs and maximizes aircraft utilization.
- o Ensure that the system can be proof-tested prior to industry development.

Task III - Identify Airframe Coordination Requirements

The electronic control for the Prop-Fan powered aircraft will provide effective management of the propulsion system, safety of flight, and displays of meaningful information for airplane operation. The requirements and recommendations of the airframe manufacturer will be incorporated in the final design of the control system.

Task IV - Research and Technology Plan

A detailed Research and Technology plan will be prepared to verify key technologies identified in this study and to ensure timely response to issues that could delay certification of a Prop-Fan powered aircraft.

SECTION 5.0
CONCLUSIONS AND RECOMMENDATIONS

SECTION 5.0

CONCLUSIONS AND RECOMMENDATIONS

The Advanced Prop-Fan Engine Technology Definition Study was successful in achieving its objectives: (1) a turboprop propulsion system to be combined with the Prop-Fan was defined; (2) the Prop-Fan propulsion system was compared to a similar technology turbofan system in a 120-passenger short-range aircraft; (3) key propulsion system technologies were identified. These technologies must be verified before initiating Prop-Fan design and development, leading to airplane certification.

The Prop-Fan propulsion system has the potential to provide very large fuel burn and direct operating cost advantages relative to a turbofan system with comparable technology. However, many key technologies have to be verified before industry will risk committing billions of dollars to the certification process for a Prop-Fan powered aircraft. In addition to the planned Large Advanced Propeller (LAP) and Propeller Test Assembly (PTA) programs, it is recommended that NASA also undertake programs involving (1) a large-size reduction gear, (2) Prop-Fan/nacelle/inlet/compressor interactions, and (3) small-size high-pressure compressors. Details of these recommended programs have been included in Section 4.5.

In addition, many engine/airframe issues remain unresolved, which could result in identification of other key technologies requiring verification. Joint engine/aircraft studies are recommended to address these key issues. These studies include: (1) free vs non-free power turbine engine/aircraft integration, (2) engine and aircraft oil heat rejection, (3) propulsion system mounting and the effect of the mounting system on vibration isolation and wing flutter, and (4) an integrated propulsion system/aircraft control.

LIST OF ABBREVIATIONS AND SYMBOLS

APET	-	Advanced Prop-Fan Engine Technology
BSFC	-	block specific fuel consumption
Btu	-	British thermal unit
C_V	-	velocity coefficient
CET	-	combustor exit temperature
CIT	-	combustor inlet temperature
CO	-	carbon monoxide
dB	-	decibel
DN	-	bearing life (diameter X speed)
DOC	-	direct operating cost
E^3	-	Energy Efficient Engine
EPNL	-	effective perceived noise level
FAA	-	Federal Aviation Administration
FAR	-	Federal Air Regulation
$\bar{F}n$	-	net thrust
ft/sec	-	feet per second
HC	-	hydrocarbon
HP	-	horsepower
HPC	-	high-pressure compressor
HPT	-	high-pressure temperature
ICAC	-	initial cruise altitude capabilities
ICAO	-	International Civil Aviation Organization
IPT	-	intermediate pressure turbine
kg/sec	-	kilogram per second
kw/min	-	kilowatt per minute
LAP	-	large advance propeller
lb/sec	-	pound per second
LCF	-	low cycle fatigue
LPC	-	low-pressure compressor
M or Mn	-	mach number
MCL	-	maximum climb
MCR	-	maximum cruise
m/sec	-	millimeter per second
MTBR	-	mean time before removal
NASA	-	National Aeronautics and Space Administration
NO_x	-	oxides of nitrogen

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LIST OF ABBREVIATIONS AND SYMBOLS (Continued)

OEW	-	operating empty weight
OPR	-	overall pressure ratio
P/P	-	pressure ratio
P/T	-	total pressure
PTA	-	propeller test assembly
SFC	-	specific fuel consumption
shp	-	shaft horesepower
STF	-	study turbofan
STS	-	study turboshaft
T_{max}	-	maximum temperature
TOFL	-	takeoff field length
TOGW	-	takeoff gross weight
TSFC	-	thrust specific fuel consumption
UTRC	-	United Technologies Research Center
V_{je}	-	jet velocity
VAMP	-	Vehicle Analysis Modular Programming System
WAF	-	fan airflow
Wa	-	airflow
θ	-	mean turning angle
δ	-	corrected pressure
Δ	-	finite change

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NASA Dryden Flight Research Center
P. O. Box 273
Edwards, CA 93523
Attn: R.S. Baron, MS D-FP (1)

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Wright Patterson AFB, OH 45433
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Allison Gas Turbine Operations
General Motors Corporation
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Attn: R.D. Anderson, MS T-18 (1)

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Attn: D.A. Wagner, MS T-18 (1)

Beech Aircraft Corporation
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Attn: R.W. Awker (2)

Boeing Commercial Airplane Company
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Seattle, WA 98124
Attn: G.P. Evelyn, MS 72-27 (3)

Boeing Military Airplane Company
P. O. Box 7730
Wichita, KS 67277-7730
Attn: D. Axelson, MS K77-24 (2)

Boeing Military Airplane Company
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Attn: C. T. Havey, MS 75-76 (2)

Cessna Aircraft Company
P. O. Box 154
Wichita, KS 67201
Attn: Dave Ellis, Dept. 178 (2)

Douglas Aircraft Co.
3855 Lakewood Blvd.
Long Beach, CA 90801
Attn: R.F. Chapier, MS 3641 (1)

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Attn: E.S. Johnson, MS 3641 (1)

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3855 Lakewood Blvd.
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Attn: F.C. Newton, MS 3584 (1)

The Garret Corporation
One First National Plaza
Suite 1900
Dayton, OH 45402
Attn: A.E. Hause (1)

General Electric Company
Aircraft Engine Business Group
1000 Western Avenue
Lynn, MA 01905
Attn: R.J. Willis, Jr., M.S. WL 345
(2)

General Electric Company
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One Neumann Way
Cincinnati, OH 45215
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Grumman Aerospace Corporation
Bethpage, NY 11714
Attn: N.F. Dannenhoffer, M.S. C32-05
(1)

Grumman Aerospace Corporation
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Attn: C. Lehman, M.S. C42-05 (1)

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Attn: J. Karanik, M.S. C32-05 (1)

Hamilton Standard Division, UTC
Windsor Locks, CT 06096
Attn: J.A. Baum, M.S. 1-2-11 (1)

Hamilton Standard Division, UTC
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Attn: S. H. Cohen, M.S. 1-2-11 (1)

Hamilton Standard Division, UTC
Windsor Locks, CT 06096
Attn: B. S. Gatzen, M.S. 1-2-11 (2)

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Hartzell Propeller Products
P. O. Box 1458
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Attn: A. R. Disbrow (1)

Lockheed-California Company
P. O. Box 551
Burbank, CA 91503
Attn: A.R. Yackle,
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86 South Cobb Drive
Marietta, GA 30063
Attn: W.E. Arndt, M.S. D/72-17,
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Attn: A. H. McKibbin, M.S. 163-12 (1)

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Sikorsky Aircraft
Transmission Engineering
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Stratford, CT 06601
Attn: R. Stone, M.S. S-318A (3)

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Walled Lake, MI 48088
Attn: Edward Lays, M.S. 4-9 (1)

Air Canada
Dorval Base H4Y-1CZ
Quebec, Canada
Attn: Goeff Haigh - Zip 14 (1)

Air Canada
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Air Transport Association
1709 New York Avenue, NW
Washington, DC 20006
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Delta Air Lines Inc.
Hartsfield Atlanta Intn'l Airport
Atlanta, GA 30320
Attn: J.T. Davis, Engr Dept. (2)

Federal Express
P. O. Box 727-4021
Memphis, TN 38194
Attn: B. M. Dotson, M.S. 4021 (1)

Ozark Air Lines Inc.
P. O. Box 10007
Lambert St. Louis Airport
St. Louis, MO 63145
Attn: Phil Rogers, Engrg Dept. (1)

Trans World Airlines Inc.
605 Third Avenue
New York, NY 10016
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United Air Lines
San Francisco International Airport
San Francisco, CA 94128
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